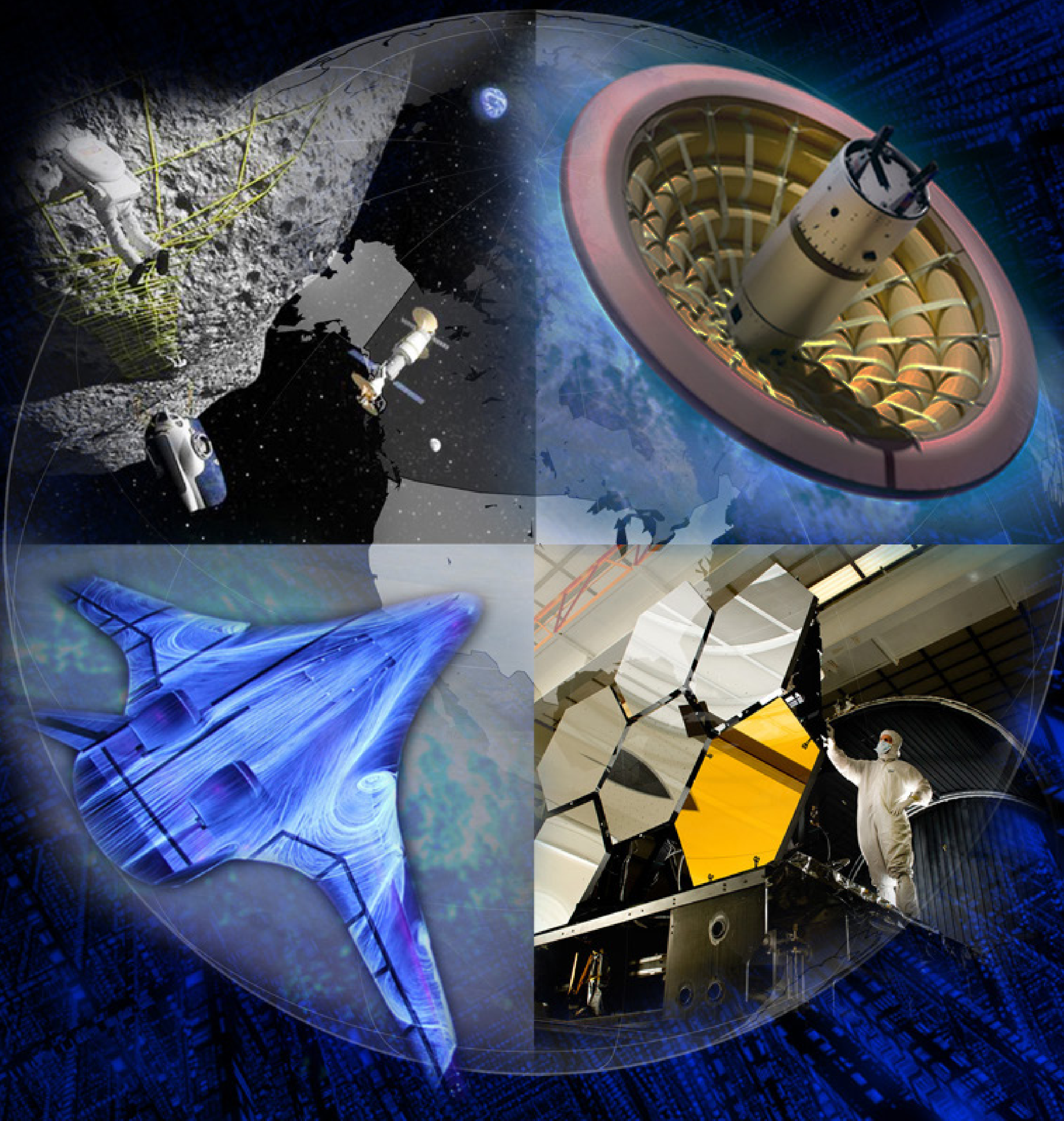
National Aeronautics and Space Administration

NASA Technology Roadmaps

TA 14: Thermal Management Systems



July 2015

Foreword

NASA is leading the way with a balanced program of space exploration, aeronautics, and science research. Success in executing NASA's ambitious aeronautics activities and space missions requires solutions to difficult technical challenges that build on proven capabilities and require the development of new capabilities. These new capabilities arise from the development of novel cutting-edge technologies.

The promising new technology candidates that will help NASA achieve our extraordinary missions are identified in our Technology Roadmaps. The roadmaps are a set of documents that consider a wide range of needed technology candidates and development pathways for the next 20 years. The roadmaps are a foundational element of the Strategic Technology Investment Plan (STIP), an actionable plan that lays out the strategy for developing those technologies essential to the pursuit of NASA's mission and achievement of National goals. The STIP provides prioritization of the technology candidates within the roadmaps and guiding principles for technology investment. The recommendations provided by the National Research Council heavily influence NASA's technology prioritization.

NASA's technology investments are tracked and analyzed in TechPort, a web-based software system that serves as NASA's integrated technology data source and decision support tool. Together, the roadmaps, the STIP, and TechPort provide NASA the ability to manage the technology portfolio in a new way, aligning mission directorate technology investments to minimize duplication, and lower cost while providing critical capabilities that support missions, commercial industry, and longer-term National needs.

The 2015 NASA Technology Roadmaps are comprised of 16 sections: The Introduction, Crosscutting Technologies, and Index; and 15 distinct Technology Area (TA) roadmaps. Crosscutting technology areas, such as, but not limited to, avionics, autonomy, information technology, radiation, and space weather span across multiple sections. The introduction provides a description of the crosscutting technologies, and a list of the technology candidates in each section.

Table of Contents

Executive Summary

This is Technology Area (TA) 14: Thermal Management Systems, one of the 16 sections of the 2015 NASA Technology Roadmaps. The Roadmaps are a set of documents that consider a wide range of needed technologies and development pathways for the next 20 years (2015-2035). The roadmaps focus on “applied research” and “development” activities.

The Thermal Management Systems technology area (TA) crosscuts and is an enabler for most other system-level TAs. As such, the design of thermal management systems inherently requires that nearly all other spacecraft systems for both human-based and robotic spacecraft be considered. Technology development in TA 14 is centered on systems with reduced mass and/or enhanced performance. Increased reliability and survivability in hostile environments are also critical goals.

Thermal management systems acquire, transport, and reject heat, as well as insulate and control the flow of heat to maintain temperatures within the specified limits. Virtually all spacecraft and related equipment require some level of thermal control, some much more tightly controlled than others, and the design approach and technologies employed vary widely depending on application. Additionally, from a thermal perspective, spaceflight hardware is highly coupled to its radiative environment per the basic laws of physics such as the Stefan–Boltzmann law. A spacecraft’s radiative environment generally varies over time, and for planetary applications, the ambient atmosphere may have a significant impact. Environments may be corrosive, abrasive, high pressure, or high temperature.

Technologies within TA 14 are organized into the three traditional sub-areas of Cryogenic Systems, Thermal Control Systems, and Thermal Protection Systems. Each of these sub-areas has unique design drivers, devices, materials, test facilities, and analytical techniques. This document addresses such technologies and is focused on the new or improved ones needed to meet NASA’s future mission requirements. Performance goals are generally very specific to each technology.

Goals

The most fundamental goal of a thermal management system is to maintain temperatures of a sensor, component, instrument, spacecraft, or space facility within the required temperature limits, regardless of the external environment or the thermal loads imposed from operations. This goal applies to all three sub-areas, but in many cases the details are dramatically different.

Cryogenic thermal management objectives center on mass and energy efficiency of components such as advanced insulation, cryocoolers, pumps, and other unique hardware. Also important is the development of large-capacity liquefaction cycles (for example, low temperature radiators for pre-cooling gas, high-performance or high-capacity cryocoolers, and two-phase flow radiators that serve as passive liquefiers) optimized for the given environment.

Thermal control systems, operating near-room temperature, maintain all vehicle surfaces and components within an appropriate temperature range throughout the many mission phases (for example, ground, launch, deployment, normal and contingency operations) despite changing heat loads and thermal environments. An effective mid-temperature thermal control system must provide three basic functions to the vehicle or system design: heat acquisition, heat transport, and heat rejection, while being mindful of the operational environment and spacecraft systems. Technology advances for near-room-temperature applications center on advanced two-phase loops, variable-heat rejection radiators, crew-safe heat transfer fluids, heat pumps, and specialized materials. Low specific mass is a key parameter.

Thermal protection technology consists of materials and systems designed to protect spacecraft from extreme high temperatures and heating during all mission phases. There is a need to advance the state of the art in several areas of thermal protection, including development of thermal protection system materials and sensors capable of surviving return from beyond low Earth orbit. Additional focus is required to replace thermal protection system materials due to obsolescence. On-orbit thermal protection system repair is a key capability for future human missions.

Meeting these goals will require a combination of imagination, creative design, enhanced computational tools, new materials, rigorous experimental techniques, and demonstration in relevant environments.

Table 1. Summary of Level 2 TAs

14.0 Thermal Management Systems	Goals:	Maintain temperatures of a sensor, component, instrument, spacecraft, or space facility within the required limits, regardless of the external environment or the thermal loads imposed from operations.
14.1 Cryogenic Systems	Sub-Goals:	Maintain cryogenic temperatures to enable longer-duration missions that use cryogenic propellants and advance the development of cryocoolers.
14.2 Thermal Control Systems	Sub-Goals:	Maintain all vehicle surfaces and components within an appropriate temperature range throughout the many mission phases despite changing heat loads and thermal environments.
14.3 Thermal Protection Systems	Sub-Goals:	Protect spacecraft and systems during ascent through, or entry into an atmosphere.

Benefits

The primary benefits from investing in thermal management technologies enabling future human- and science-based missions are increasing system safety and performance, adding new capabilities, reducing system mass, and increasing system reliability. These technologies are crucial for conserving cryogenic fluids for propulsion, maintaining critical life support for human missions, enabling proper thermal control for sensors and instruments used in science missions, and protecting spacecraft systems from extreme thermal environments such as those encountered during atmospheric reentry.

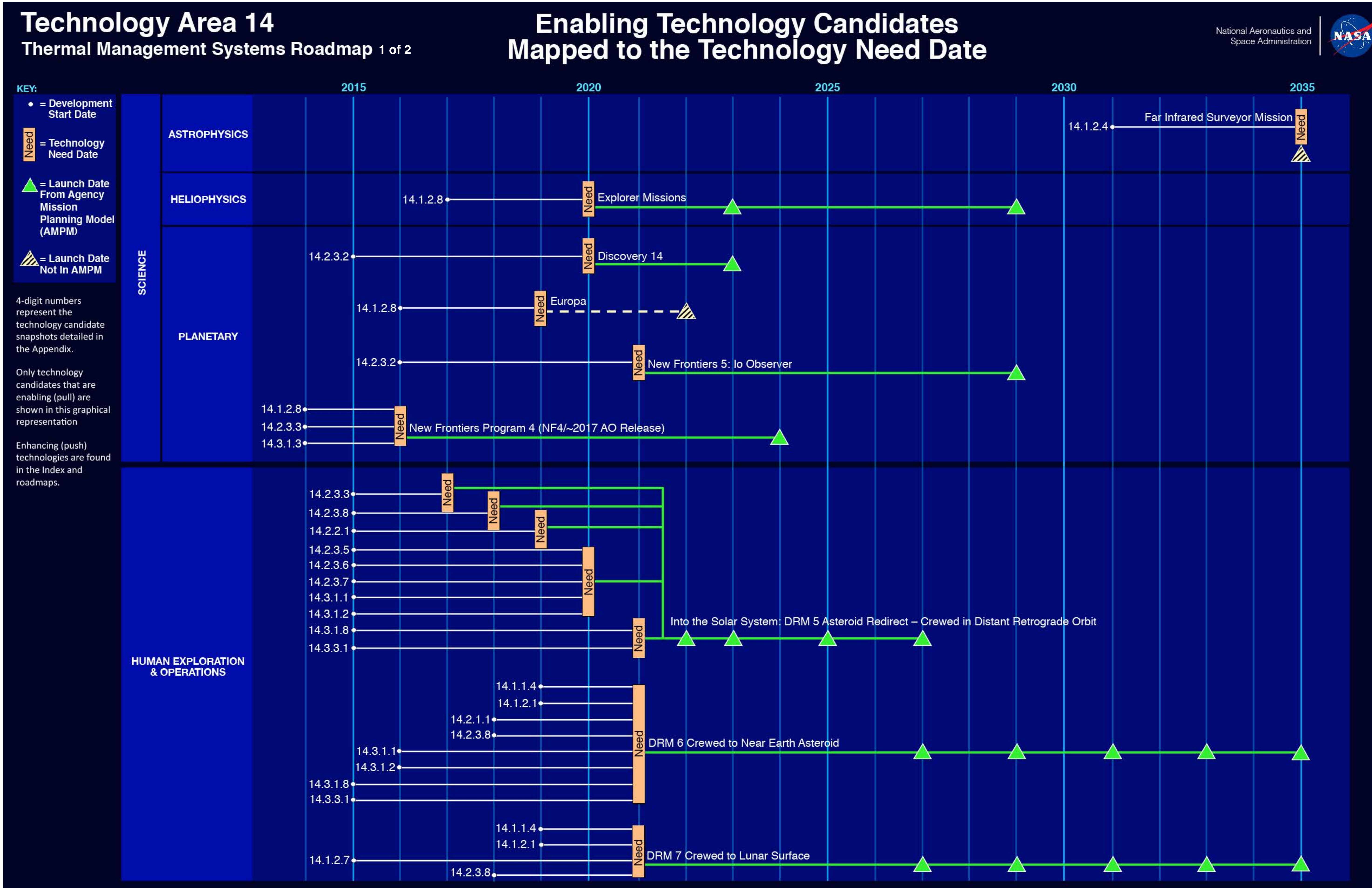


Figure 1. Technology Area Strategic Roadmap

Technology Area 14

Thermal Management Systems Roadmap 2 of 2

Enabling Technology Candidates Mapped to the Technology Need Date

National Aeronautics and Space Administration

KEY:

- • = Development Start Date
- Need = Technology Need Date
- ▲ = Launch Date From Agency Mission Planning Model (AMPM)
- ▲ = Launch Date Not In AMPM

4-digit numbers represent the technology candidate snapshots detailed in the Appendix.

Only technology candidates that are enabling (pull) are shown in this graphical representation

Enhancing (push) technologies are found in the Index and roadmaps.

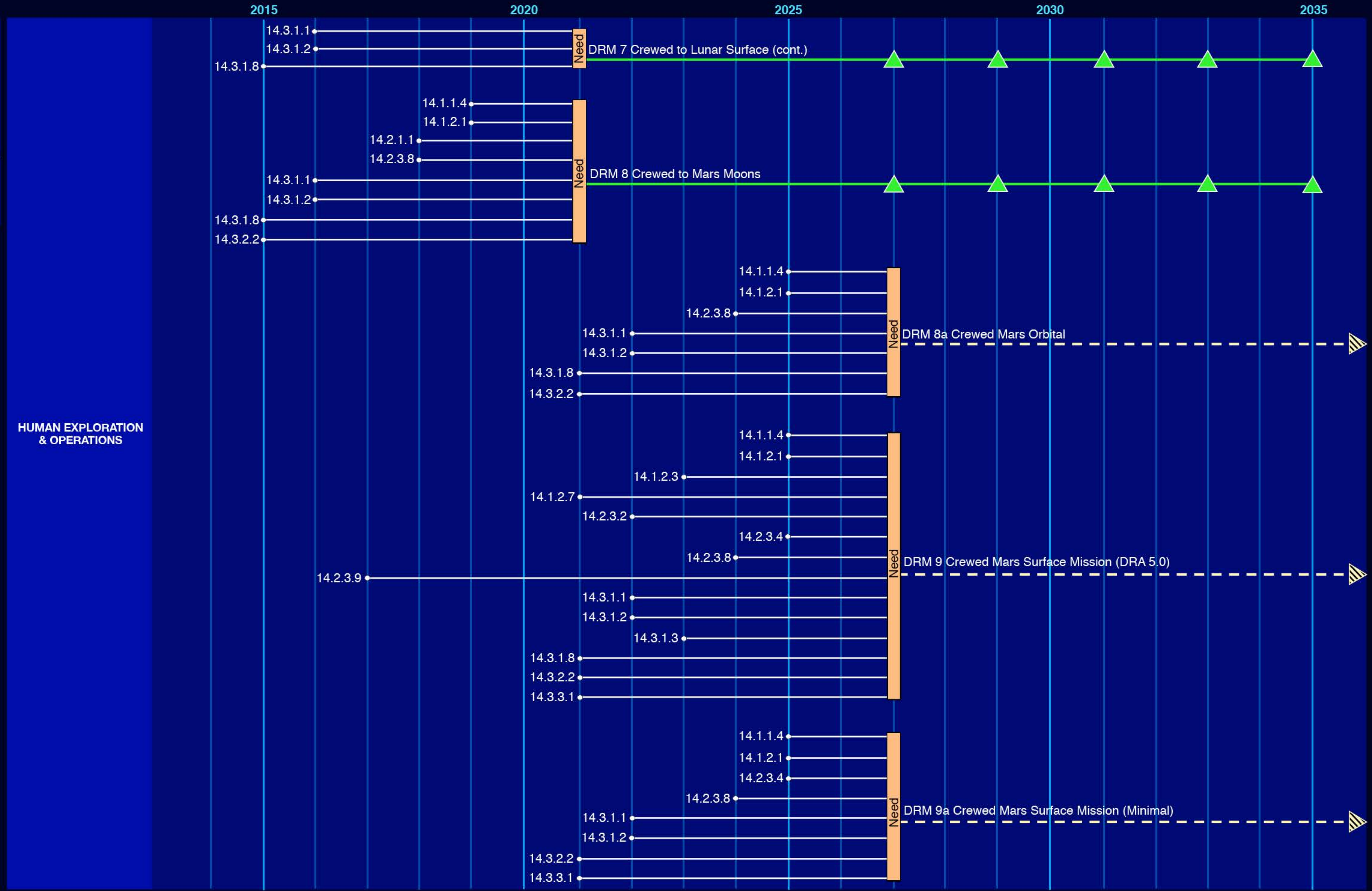


Figure 1. Technology Area Strategic Roadmap (Continued)

Introduction

Figure 2 shows the Technology Area Breakdown Structure (TABS) for the Thermal Management Systems technology area (TA). The technology is divided into three major areas: cryogenic systems, thermal control systems, and thermal protection systems. These three areas are described below.

TA 14 **Thermal Management Systems**

14.1 Cryogenic Systems
- 14.1.1 Passive Thermal Control
- 14.1.2 Active Thermal Control
- 14.1.3 Integration and Modeling

14.2 Thermal Control Systems
- 14.2.1 Heat Acquisition
- 14.2.2 Heat Transport
- 14.2.3 Heat Rejection and Energy Storage

14.3 Thermal Protection Systems
- 14.3.1 Ascent/Entry TPS
- 14.3.2 TPS Modeling and Simulation
- 14.3.3 TPS Sensors and Measurement Systems

Figure 2. Technology Area Breakdown Structure for Thermal Management Systems

NASA's technology area breakdown structure (TABS) is in wide use in technology organizations around the globe. Because of this, any sections that were previously in the structure have not been removed, although some new areas have been added. Within these roadmaps, there were some sections of the TABS with no identified technology candidates. This is either because no technologies were identified which coupled with NASA's mission needs (either push or pull) within the next 20 years, or because the technologies which were previously in this section are now being addressed elsewhere in the roadmaps. These sections are noted in gray above and are explained in more detail within the write-up for this roadmap.

14.1 Cryogenic Systems

Cryogenic systems are those systems that operate below -150 degrees Celsius (° C) or 123 Kelvin. These include advanced insulation, cryocoolers for both scientific and propellant storage applications, and various heat transfer and insulators or specialized thermal isolation techniques. In TA 14, only the thermal control aspects of cryogenic systems are addressed. Technologies related to other areas, such as liquid cryogen transfer, in-situ production, and mass gauging, are covered in other NASA technology roadmaps.

Thermal control technologies for cryogenic applications are divided into either passive or active systems.

- 14.1.1 Passive Thermal Control: Technologies for passive thermal control systems for cryogenic storage and handling include systems or components that do not require power or energy to operate or perform the desired function. In general, the objective is to maximize the passive cooling or insulating capabilities with adequate structural properties and minimum system mass. Advances in materials will influence many of these technologies and, where possible, should be integrated with dual functionality, such as protection from micrometeorites. This area includes insulations for various stages of the mission, low-conductivity structures and supports, radiators, and heat pipes.

- 14.1.2 Active Thermal Control: Technologies for active thermal control systems for cryogenic storage and handling include systems or components that require power or energy to operate or perform the desired function. In general, the objective is to reduce or eliminate boil-off of propellants to enable long-term storage. The thermal requirements are dependent on the specific fluid. Other technologies enable cooling for scientific instruments and in-situ processing of regolith or atmosphere gases into cryogens such as oxygen.
- 14.1.3 Integration and Modeling: Integration and modeling is now addressed in sections 14.1 and 14.3.

14.2 Thermal Control Systems

Mid-temperature Thermal Control Systems are those systems that are operating between -150 °C and 500 °C. These systems are broken down by the functions of heat acquisition, heat transfer, and heat rejection and energy storage:

- 14.2.1 Heat Acquisition: Heat acquisition is the process of acquiring excess thermal energy from various spacecraft components, including power systems, electronics, avionics, computers, and metabolic loads from crew members. Acquisition systems generally transfer the collected heat to a thermal transport loop or thermal sink.
- 14.2.2 Heat Transport: Heat transport is the process of transporting the acquired energy, via a thermal fluid transport loop or conductive materials, to a centralized location to be rejected or stored.
- 14.2.3 Heat Rejection and Energy Storage: Heat rejection is the process of disposing of the acquired energy, now “waste heat”, to maintain spacecraft systems within the required temperature limits. Heat rejection is generally accomplished by rejecting waste energy to the space environment via radiation, or by sublimating or evaporating a consumable, such as water or ammonia. Sublimation and/or evaporation may also be used to extend the life of spacecraft/instruments in an extreme environment such as Venus or other planetary surfaces. The acquired energy may also be stored for later rejection when environmental conditions are more favorable.

14.3 Thermal Protection Systems

Thermal Protection Systems (TPS) are thermal management systems that operate above approximately 100 °C, although often much higher. TPS technologies are further organized into three sub-categories: ascent/entry TPS, TPS modeling and simulation, and TPS sensors and measurement systems:

- 14.3.1 Ascent/Entry TPS: Ascent/entry TPS encompasses rigid ablative and reusable TPS, seals and thermal barriers, and materials and systems development driven by both materials obsolescence and newer technologies like self-repairing TPS, health monitoring, in-space TPS repair, and flexible and deployable systems.
- 14.3.2 Thermal Protection System Modeling and Simulation: TPS design and analysis is multi-disciplinary in nature. After more than 50 years of spaceflight, analytical tools that can accurately predict TPS environments, as well as the thermal, chemical, and structural response of the TPS components, are required. Modeling and simulation are critical to the TPS technology area, as current design challenges are still driving tool development to higher fidelity, multi-physics predictive capabilities that reduce the need for excessive, and in some cases prohibitive, sizing margins and flight tests.
- 14.3.3 Thermal Protection System Sensors and Measurement Systems: Sensors and measurement systems are needed for TPS development and operation. Current design challenges drive material

development and system architecture advancement to provide capabilities for higher speed reentry and larger payloads. The resulting reentry environments include effects that are not as well understood and require not just improved analytical tools, but instrumentation that can provide test and flight data for validation of the model against the real-world environment.

TA 14.1: Cryogenic Systems

Virtually any large-scale space mission, robotic or crewed, requires the use of cryogen propellants because of their high energy and performance. Use of liquid oxygen/liquid hydrogen systems have the highest thrust per mass flow rate, or specific impulse (I_{sp}), of any chemical propulsion technologies and have been used for many years in vehicles, including the space shuttle. However, these systems rapidly consume their propellant, typically in less than 24 hours after launch, so current thermal management and insulation systems are adequate and boil-off of the propellant is acceptable. Longer-duration missions using cryogenic propellants require near-zero boil-off for all systems.

Zero boil-off for liquid oxygen has been demonstrated in large-scale ground tests, but not in a microgravity environment. For liquid hydrogen, the state of the art (SOA) evaporation rate is still on the order of two percent per day.

Table 2. Summary of Level 14.1 Sub-Goals, Objectives, Challenges, and Benefits

Level 1		
14.0 Thermal Management Systems	Goals:	Maintain temperatures of a sensor, component, instrument, spacecraft, or space facility within the required limits, regardless of the external environment or the thermal loads imposed from operations.
Level 2		
14.1 Cryogenic Systems	Sub-Goals:	Maintain cryogenic temperatures to enable longer-duration missions that use cryogenic propellants and advance the development of cryocoolers.
Level 3		
14.1.1 Passive Thermal Control	Objectives:	Maximize the passive cooling or insulating capabilities with adequate structural properties and minimum system mass.
	Challenges:	Advanced materials, integrated with dual functionality such as protection from micrometeorites.
	Benefits:	Reduces mass and provides higher performance with greater simplicity.
14.1.2 Active Thermal Control	Objectives:	Reduce or eliminate boil-off of propellants and advance cryocooler technology.
	Challenges:	Improved cryocooler technologies, distributive cooling loops and hardware components for mixing cryogen fluids.
	Benefits:	When integrated with passive technologies, achieves zero boil-off for long-duration missions and improves performance of scientific instruments.
14.1.3 Integration and Modeling	This section has been incorporated into TA 14.1 and 14.3.	

TA 14.1.1 Passive Thermal Control

This technology area is comprised of eight types of passive thermal control technologies, which are:

Load Responsive Insulation: A single insulation solution that is responsive to thermal loads from launch (one atmosphere), ascent, and on orbit (vacuum) is required. Current spray-on foam insulation is optimized for pre-launch and ascent, but essentially adds dead mass once in orbit. The possibility also exists to expand multi-layer insulation (MLI) concepts so the outer layer is capable of supporting a soft vacuum while on Earth, compressing slightly while being supported by an advanced spacer system.

Wrapped Insulation: Develop high performance insulation that can be used for cryogen plumbing and components in space. The SOA is to wrap plumbing and components in MLI, but new technologies with heat transfer less than 0.25 W/m are needed for long-duration missions.

Insulation with Micrometeoroid Orbital Debris (MMOD) Protection: New MLI concepts have been proposed that eliminate the need for low-conductivity layers of paper between the radiation shields. Analysis and optimization of MLI systems to increase MMOD protection are needed. Self-healing materials that can repair damage from handling or micrometeoroids while maintaining thermal performance should be investigated. These self-healing systems are perhaps 15 to 20 years in the future.

Cooled Insulation for Reduced and Zero Boil-Off: Large-scale insulation that can be integrated with broad area coolers or hydrogen (H_2) vapor-cooled shields for large diameter tanks are required.

Modeling for Multi-layer Insulation: Multi-layer insulation heat loss is typically predicted using empirical equations developed in 1974. The accuracy of heat loss per unit area is 2.5 to 5 times lower than predicted.

Low Thermal Conductivity Structural Supports: Conduction heat loss across mechanical supports such as struts, skirts, and feedlines can be greater than the convection and radiation heat loss across the tank surface. Innovative methods for minimizing or eliminating this loss are needed. Further enhancements can be made to intercept conduction heat loss at a higher temperature by actively cooling or vapor cooling these solid structures. The optimal long-term solution is a structure that is part of the load path during ground and launch phases but disconnects on orbit. Cryogenic couplings with an autonomous docking mechanism will be necessary for cryogenic depot architecture as well.

Low-Temperature Radiators: Low-temperature radiators require very large areas because they are rejecting heat at a very low temperature. Presently, the low temperature limit is approximately 50 Kelvin (K) for low-Earth orbit (LEO) applications, and this is for very small loads (milliwatt scale). Advances like deployable systems are needed so they are packaged for launch and deployed on orbit with a much larger radiative area. Ideal materials are those that are flexible for launch but are capable of being made rigid in space while maintaining very high emissivity.

Cryogenic Heat Pipes: Heat pipes and heat spreaders that are effective at temperatures below 50 K are required. Specialized applications require devices that operate below 4 K for scientific instruments. Since most fluids freeze at these temperatures, there is a need for helium, hydrogen, and neon heat pipes capable of providing cooling transport at 4 K, 15 K and 40 K, respectively. Due to fundamental material properties, there is a lack of working fluids for 10 K two-phase loops. For example, helium operates below 6 K, hydrogen operates above 14 K, but no fluids operate in between. Cryogenic heat pipes are also addressed in the technology roadmap for TA 8, Science Instruments, Observatories, and Sensor Systems.

Technical Capability Objectives and Challenges

Passive thermal control technologies for cryogenic systems require much advancement. Current multi-layer insulation (MLI) concepts utilize low thermal conductivity layers of paper between each radiation layer for spacing but add mass and reduce performance. Effective ground-to-flight insulation and MLI with micrometeoroid and orbital debris (MMOD) protection have not been demonstrated.

In general, the objective is to maximize the passive cooling or insulating capabilities with adequate structural properties and minimum system mass. Advances in materials development will drive many of these technologies and, where possible, should be integrated with dual functionality, such as protection from micrometeorites. This area includes insulations for various stages of the mission, low conductivity structures and supports, radiators, and heat pipes.

Benefits of Technology

Lower mass and higher performance are predicted benefits over current systems. Integration of multifunctional insulating materials into other spacecraft systems can significantly reduce spacecraft mass and increase simplicity. For instance, MLI has shown some ability to serve as an effective MMOD protection while enabling cryogenic propellant systems.

Table 3. TA 14.1.1 Technology Candidates – not in priority order

TA	Technology Name	Description
14.1.1.1	Load-Responsive Insulation	Multi-environment thermal insulation for spacecraft cryogenic propellant tanks.
14.1.1.2	Wrapped Insulation	Thermal insulation for cryogenic tubing.
14.1.1.3	Insulation with Micrometeroid and Orbital Debris Protection	Spacecraft cryogenic propellant-tank thermal insulation with micrometeoroid and orbital debris protection.
14.1.1.4	Cooled Insulation for Reduced and Zero Boil-off	Insulation integrated with broad area cooling (BAC) shields or hydrogen vapor-cooled shields.
14.1.1.5	Modeling for Multi-Layer Insulation	Empirical equations for low-temperature multi-layer insulation.
14.1.1.6	Low Thermal Conductivity Structural Supports	Structural supports for cryogenic propellant tanks that have low heat loss.
14.1.1.7	Low-Temperature Radiators	Spacecraft radiators that have high heat rejection at very low temperatures.
14.1.1.8	Cryogenic Heat Pipes	Heat pipes and heat spreaders that are effective at temperatures below 50 K. Specialized applications require devices that operate below 4 K.

TA 14.1.2 Active Thermal Control

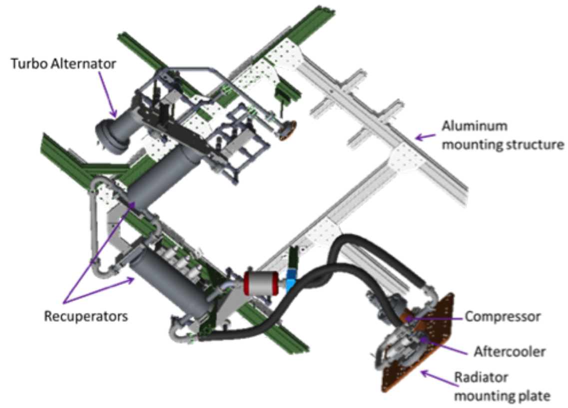
This technology area is comprised of eight types of active thermal control technologies, which are:

High Capacity 20 K Cryocoolers: Includes development of cryocoolers with a cooling capacity of greater than 20 Watts (W) (at 20 K); a specific power of less than 80 Watts per Watt (W/W); and a specific mass of less than 5 kg/W. This technology will enable long-duration storage of cryogenic propellants in orbit. Our current propulsion stage experience threshold is about nine hours' time on orbit prior to reentry, but will need to be extended to two to three years for some future missions. Applications include on-orbit cryogenic depots, long-duration Mars stages, and in-situ resource utilization (ISRU) production on lunar or Martian surfaces.

High Capacity 90 K Cryocoolers: Includes development of cryocoolers with a cooling capacity of greater than 150 W (at 90 K); a specific power of less than 10.6 W/W; and a specific mass of less than 0.35 kg/W. High-capacity 90 K cryocoolers to achieve zero boil-off for liquid oxygen or liquid methane propellant are required. In addition, they can be used to reduce boil-off for liquid hydrogen and reduce the required cooling capacity of the 20 K cryocoolers.

High Capacity Cryocoolers for In-Situ Manufacture of Cryogenic Fluids: Includes development of cryocoolers with the ability to capture, purify, and compress Martian atmospheric gases for processing at a rate of 12.1 kg CO_2/hr (to produce 2.2 kg O_2/hr). Eventual human bases on the Moon or Mars will rely on ISRU to produce necessary propellants and life-support consumables. For propellant use, liquefaction will be required. The SOA is large-scale liquefaction and production plants on Earth. Development of large-capacity liquefaction cycles that are optimized for the given environment will be important. This includes low-temperature radiators for pre-cooling gas, as well as potential two-phase flow radiators that serve as passive liquefiers. Integration of the cooling cycle with the ISRU plant requires a thermal system, including effective recuperator heat exchangers and high-pressure electrolysis systems that serve as compressors for the liquefaction cycle.

Low T, Low Q Cryocoolers: Includes development of low-power 35 K, 10-6 K, and 2 K cryocoolers to cool the next generation of science instruments. 35 K coolers are needed for mercury cadmium telluride long wave infrared (IR) detectors. 10 K – 6 K coolers are needed for arsenic-doped silicon detectors, which operate in the IR spectrum. 2 K coolers are needed as upper stage (that is, the cryocooler) for lower stage, adiabatic demagnetization refrigeration (ADR) systems for x-ray spectrometers. Advances in magnetic materials for ADRs to increase the temperature regime, including high-temperature superconductors, could offer alternatives to current cryocooler systems at 20 to 70 K. Higher-density magnetic materials are needed. New paramagnetic materials are needed for ADR operation at 20 milliKelvin (mK). This topic is also addressed in the technology roadmap for TA 8, Science Instruments, Observatories, and Sensor Systems.



Creare Reverse Turbo-Brayton Cycle Cryocooler

Distributed Cooling Loops: There is a requirement for loops to distribute cooling boil-off vapor or circulated cryogenic gas over large surfaces within the insulation and to discrete locations, such as piping or structural supports that penetrate the insulation.

Pumps, Circulators, and Fans: Development of cold-gas compressors and cold-liquid-circulating pumps with long life, variable speed operation, and very low leakage is needed for intermediate heat transport loops. Cold control valves with low heat loss are also needed. Piezoelectric materials have potential to offer lower heat loss from actuators. Advances in compressor design using non-contact bearings such as magnetic or foil configurations will increase system reliability of these components. Reliable cryo-valves for scientific instruments are also an area needing development. These technologies are important for cryogen processing and sub-cooling.

Integrated Radiator/Cryocooler for Liquefaction: An integrated radiator/cryocooler system for continuous liquefaction product stream at an in-situ processing plant is required. This includes a high capacity, high efficiency, low mass heat rejection system and controls for oxygen production on the Martian surface.

Subcooling Cryogenic Propellants: Subcooling cryogenic propellants prior to launch to prevent boil-off for extended durations is required. For interplanetary missions, subcooling can provide years of vent-free hydrogen storage without additional launched mass, enabling the use of high specific impulse liquid hydrogen and liquid oxygen engines.

Technical Capability Objectives and Challenges

The SOA for cryocoolers is approximately 1 W at 20 K with a specific power of 200 W/W. For higher temperatures, the cooling capacity increases (for example, 6W at 50 K). To achieve zero boil-off of liquid hydrogen, cryocooler technology requires significant advancement, on the order of 20W at 20 K. In addition, the reliability of cryocoolers must increase significantly for missions lasting more than 18 months. Development of the recuperator fabrication process and integration of the recuperator, compressor, and turbo-alternator are also challenges that must be addressed.

In general, the objective is to reduce or eliminate boil-off of propellants to enable long-term storage. The thermal requirements are dependent on the specific fluid (for example, liquid oxygen, liquid hydrogen, or liquid methane). Other technologies enable cooling for scientific instruments and in-situ processing of regolith or atmospheric gases into cryogens, such as oxygen. There is a severe shortage of understanding and modeling capability (even at the computational fluid dynamics level) of cryogenic fluids in the two-phase state. Such modeling should be included in the appropriate hardware development efforts, and is also referenced in the technology roadmap for TA 8, Science Instruments, Observatories, and Sensor Systems.

Benefits of Technology

Active thermal control technologies for cryogen fluids can be integrated with passive technologies to achieve near zero boil-off for long-duration missions. High capacity or very low temperature cryocoolers are also required for in-situ manufacturing of cryogenic fluids or for specialized science instruments. For launch vehicle upper stages, subcooling can provide weeks to months of in-space, vent-free hydrogen storage (depending on parking orbits) without additional launched mass, allowing fewer launches and less massive launch vehicles.

Table 4. TA 14.1.2 Technology Candidates – not in priority order

TA	Technology Name	Description
14.1.2.1	High-Capacity 20 Kelvin Cryocoolers	Spaceflight cryocooler for cooling liquid hydrogen propellant tanks.
14.1.2.2	High-Capacity 90 Kelvin Cryocoolers	Spaceflight cryocooler for cooling liquid oxygen propellant tanks and for broad-area cooling shield for liquid hydrogen tanks.
14.1.2.3	High-Capacity Cryocoolers for In-Situ Manufacture of Cryogenic Fluids	High capacity, high efficiency, low mass cryocoolers for liquid oxygen production on the Martian surface.
14.1.2.4	Low-T, Low-Q Cryocoolers	Multi-stage, sub-Kelvin cryocoolers with high efficiency, and 4 K cryocoolers as heat sinks.
14.1.2.5	Distributed Cooling Loops	Cooling systems that reduce boil-off from cryogenic propellant tanks. May include cryogenic circulators, heat traps, and heat exchangers.
14.1.2.6	Pumps, Circulators, and Fans	Devices for transporting cryogenic liquids and gases are need for many applications, such as propellant-tank mixing, cryogenic fluid transfer, broad-area cooling loops, etc.
14.1.2.7	Integrated Radiator/Cryocooler for Liquefaction	High-capacity, high-efficiency, low-mass heat rejection system and controls for oxygen production on the Martian surface.
14.1.2.8	Subcooling Cryogenic Propellants	Allows extended storage time for cryogenic fluids post launch. Vent-free (on-ground) storage may also be possible with this technology.

TA 14.1.3 Integration and Modeling

Integration and Modeling is addressed in sections 14.1 and 14.3.

TA 14.2: Thermal Control Systems

The temperature regime of thermal management systems is often associated with the narrow control set-points required by human spaceflight, instruments, sensors, and the need for spacecraft thermal management across a wide range of mission thermal environments. The SOA for human spaceflight vehicles includes heat acquisition using coldplates and condensing heat exchangers, heat transport using dual-loop systems designed with non-toxic, near room-temperature internal-loop fluids, and external-loop fluids that are acceptable for the wide environment temperature ranges and heat loads experienced during the mission. Robotic spacecraft thermal control is similar, but without a concern for toxicity. In addition, some instruments and sensors may require exceptionally tight thermal control to milliKelvin stability. Science missions are also increasingly relying more on micro and nano satellites due to their lower cost and easier access to space. Hence, miniaturization of thermal control systems is a general goal.

Two-phase devices are often used to accomplish heat transport. Heat rejection is accomplished through deployable or body-mounted radiators that essentially have a fixed emissivity and view to the space environment. The combined SOA results in more massive, less reliable systems due to the complexity of the dual-loop system and the need to reject heat at maximum mission heat loads and hottest thermal environments. Advances in mid-temperature thermal control technologies will allow for single-loop transport systems for crewed missions using non-toxic fluids and the ability to “turn down” heat rejection via the radiators during cold mission environments and relatively low heat loads.

Technology development in thermal control systems promises to increase performance, reduce mass, increase safety and reliability, and provide spacecraft systems capable of operating in a broader range of mission durations and environments. The thermal control system for any defined mission will be tailored to include a combination of technologies that meet the needs of spacecraft systems, human life support, habitation infrastructure, science instruments, and logistics needs while responding to changing mission environments, spacecraft heat loads, science objectives, and operational considerations. By developing these thermal control technologies, NASA will be positioned to acquire, transport, and reject heat to meet the needs of any of the design reference missions (DRMs) it implements.

Table 5. Summary of Level 14.2 Sub-Goals, Objectives, Challenges, and Benefits

Level 1		
14.0 Thermal Management Systems	Goals:	Maintain temperatures of a sensor, component, instrument, spacecraft, or space facility within the required limits, regardless of the external environment or the thermal loads imposed from operations.
Level 2		
14.2 Thermal Control Systems	Sub-Goals:	Maintain all vehicle surfaces and components within an appropriate temperature range throughout the many mission phases despite changing heat loads and thermal environments.
Level 3		
14.2.1 Heat Acquisition	Objectives:	Reduce mass and increase efficiency of heat acquisition components.
	Challenges:	Freeze-tolerant heat pipes and high-flux heat acquisition components. High thermal conductivity, low-density materials with favorable manufacturing characteristics.
	Benefits:	Reduces mass and increases efficiency, leading to smaller and more efficient heat exchangers that do not require brazed joints. Freeze-tolerant heat pipes allow a wider range of mission thermal environments with variable heat loads. Damage-tolerant and self-healing heaters provide more reliable thermal control systems.

Table 5. Summary of Level 14.2 Sub-Goals, Objectives, Challenges, and Benefits - Continued

Level 3		
14.2.2 Heat Transport	Objectives:	Increase efficiency of heat transfer systems (fluid loops, heat pipes, insulations) and optimize transport fluids for environmental and operational design requirements.
	Challenges:	Transport fluids with performance characteristics that are suitable for human spaceflight missions, reliable fluid pumps for fluid-loop systems, efficient heat pumps to reliably move heat against a thermal gradient, passive conductors and heat pipe capillary loops that are tailored to specific applications, and high-performance and lightweight insulators that can prevent heat transport between components and between the spacecraft and the space environment.
	Benefits:	Allows safe, reliable, and reduced-mass transport loop designs for human missions. Allows more efficient and reliable systems that reduce power requirements and permits heat transport against a steeper thermal gradient, thus allowing spacecraft to operate in a wider range of environments. Provides more efficient heat switches and heat straps and insulations that provide more reliable and efficient passive thermal control of spacecraft.
14.2.3 Heat Rejection and Energy Storage	Objectives:	Provide turndown capability for heat rejection and reduce contaminant sensitivity.
	Challenges:	Vary radiator heat rejection rates to match mission phase requirements based on heat loads and thermal environments; perform real-time mission repairs of damaged radiator systems; and manage radiator surface optical properties during extended deep-space and surface missions. Reduction of contamination sensitivity and recapture of consumables. Mass reduction through use of water as opposed to wax for latent energy storage and management of freeze and thaw dynamics to assure efficient and reliable performance. Extended survival in extreme envrionments.
	Benefits:	Provides more capable heat rejection systems across a wider range of thermal environments and heat loads, optimizes design of heat rejection systems, and assures radiator performance throughout the mission. Reduces risks resulting from contamination and may lead to consumables recovery, which will result in significant mass savings. High-temperature, two-phase systems and heat pipes support high-power system requirements, such as nuclear power systems. Extended survival in extreme environments such as the surface of Venus or other planetary applications.

TA 14.2.1 Heat Acquisition

This technology area is composed of four types of heat acquisition technologies including:

Freeze-Tolerant Heat Pipes must be able to allow repeated freeze/thaw cycles of the operating fluid, typically ammonia, but other fluids are possible, without bursting or degrading performance. For ammonia-based heat pipes, by far the most common technology, it is desirable to increase freeze tolerance to temperatures below 150 K in order to meet anticipated survival requirements for a variety of deep-space missions, DRM 6 and DRM 8, and satellites at geosynchronous orbit (GEO).

High-Flux Heat Acquisition with Constant Temperature provides enhancement for many future missions where high heat flux requires tight temperature control (for example, integrated circuit chips, laser heads, and other electronic parts). Current technology for high-flux heat removal includes high flow rates of a single-phase fluid, vapor/liquid spray, or advanced heat pipes. These devices have limitations in maximum flux levels, degree of temperature control, and erosion of the surface. Also, demonstration of operation in micro-gravity is needed.

Damage-Tolerant or Self-Healing Heaters must demonstrate significantly reduced loss of functionality due to physical damage, delamination, or over-powering. This is a broad capability that would be useful for almost all spacecraft missions. New materials, possibly involving advanced nano-infused wires, and heater concepts that could increase isothermally and reliability and minimize burnout or thermal fatigue, are critical.

Insulation must meet challenges for protecting spacecraft components and systems from thermal environments and control heat transfer between components and subsystems. Advances in thermal insulation will benefit all future spaceflight missions and are considered enhancing. Further increases in insulative properties require development of new materials and insulation configurations. In addition, it is desirable to develop insulation that is less labor intensive to install.

Technical Capability Objectives and Challenges

Heat acquisition is typically accomplished using a myriad of hardware components, mainly coldplates, air/liquid heat exchangers, and liquid/liquid heat exchangers. Advances in heat exchangers and coldplate technology primarily reside with materials and manufacturing advancements and are covered in those technical areas. Heaters are also considered a component of heat acquisition and are generally used to address temperature differences created by the environment or differences in heat generation between components or areas of the spacecraft. Coldplates are used to capture heat from vehicle systems and equipment and air/liquid heat exchangers condition air and remove condensate from crew-occupied volumes. Liquid/liquid heat exchangers, typically used in crewed applications, provide an interface between interior heat acquisition pumped thermal loops and exterior heat rejection pumped thermal loops. In some situations, particularly for robotic spacecraft, heat-dissipating equipment is attached directly to structural members, which may also act to transport heat or to directly radiate waste heat to space. Heaters provide supplemental energy to systems with operating temperatures above ambient environmental temperatures, and insulation prevents systems from exchanging heat with the environment.

The primary objectives in heat acquisition technology development are increasing efficiency and reducing the mass of heat acquisition components, as well as increasing the reliability and simplicity of heater operation. As such, most of these technologies are considered enhancing for most design reference missions. Developing freeze-tolerant heat pipes and high-flux heat acquisition components will help expand the capability for heat acquisition in selected applications. Advances in materials with high thermal conductivity, low density, and favorable manufacturing characteristics are required, and are shared with TA 12, Materials, Structures, Mechanical Systems, and Manufacturing.

Coldplates and heat exchangers are thermal devices that require reductions in heat exchanger and coldplate mass with increased efficiency, which will benefit all future spaceflight missions. These technologies are considered enhancing because they offer component mass reduction as part of the active thermal control system. Higher efficiency/lower mass (W/Kg-°C) designs can be realized through the use of micro-channel fabrication techniques, additive manufacturing, or the use of composite materials. Composite materials are desirable for the fabrication of heat exchangers due to their potentially high thermal conductivity and high strength-to-mass ratio, especially composites enhanced by nanotechnology. Microchannel fabrication techniques can also be used to reduce the fin spacing by an order of magnitude, increasing the thermal performance while also dramatically reducing the hardware mass and volume. The potential for additive manufacturing techniques to improve heat transfer and mass ratios may be even larger. The development of materials and manufacturing processes for heat exchanger and coldplate development is addressed in the technology roadmap for TA 12, Materials, Structures, Mechanical Systems, and Manufacturing.

Benefits of Technology

Heat acquisition, or the minimization of heat acquisition, is a primary function of any thermal control system. Reducing component mass and increasing efficiency can provide significant mass reduction and performance increase in spacecraft insulations and thermal control heat acquisition. Advances in manufacturing techniques will lead to smaller and more efficient heat exchangers that do not require brazed joints. Freeze-tolerant heat pipes will allow a wider range of mission thermal environments with variable heat loads. Damage-tolerant and self-healing heaters will provide more reliable thermal control systems.

Table 6. TA 14.2.1 Technology Candidates – not in priority order

TA	Technology Name	Description
14.2.1.1	Freeze-Tolerant Heat Pipes	Heat transport devices that can freeze and thaw without damage or degradation in performance.
14.2.1.2	High-Flux Heat Acquisition with Constant Temperature	Micro- and nano-scale heat transfer enhancement technologies that use latent heat to maintain constant temperature.
14.2.1.3	Damage-Tolerant or Self-Healing Electric Heaters	Advanced electro-resistive materials, including nanotechnology, and component design.
14.2.1.4	Insulation	Lightweight thermal insulators with low conductivity and effective emissivity (ε*).

TA 14.2.2 Heat Transport

This technology area is composed of eleven types of heat transfer technologies, including:

Heat Transport Fluids: For human-tended spacecraft, it is desirable to develop technologies that enable single-loop thermal control system architectures, rather than the state of the art (SOA), which uses an internal/external two-loop architecture. The single-loop architecture would have benefits of improved reliability, system simplicity, and significant mass savings. Typically, thermal control systems, including the vehicle's radiator, are sized for the maximum continuous heat load in the warmest sustained thermal environment. This design approach results in a fairly large radiator surface area. Unless some sort of protection is provided, the heat transport fluid will require that a relatively low freeze temperature be incorporated into the system design to avoid freezing the fluid during cold and low-heat load mission phases. Unfortunately, most of the fluids that have low freeze temperatures such as fluorocarbons and ammonia, are high vapor pressure oxygen displacers or are toxic to crew members and cannot be used inside the pressurized spacecraft volume for fear of an inadvertent system leak and crew exposure. As a result, two-loop thermal control system architectures are typically designed to use low freeze-point temperatures in the external loop and a more benign fluid in an internal loop that is connected via an inter-loop heat exchanger, which adds substantial mass to the system. Single-loop architectures could save significant mass and increase reliability by eliminating components. They could be enabled either by developing advanced fluids or developing variable-heat-rejection radiator technologies. The development of advanced fluids should be focused on low toxicity while depressing the freeze temperature and ensuring that the advanced fluid has other thermophysical properties comparable to water.

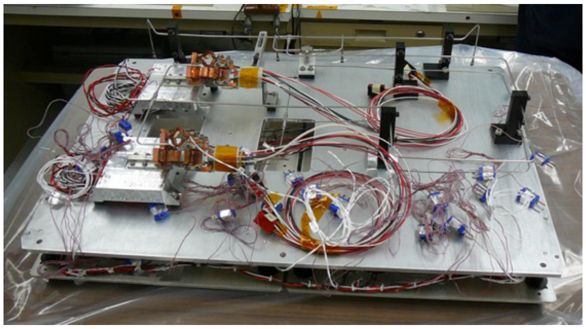
Robotic missions to outer planets will also require advanced fluids with low freeze temperatures, high thermal conductivity, and high radiation tolerance. This includes fluids for both single and two-phase applications.

Advanced Pumps are needed for both human and robotic missions. Such pumps must achieve at least several times the SOA, about 50,000 hours, mean-time-between-failure in order to meet applications such as missions to the outer planets. A sensorless, radiation-hardened pump is specifically desired. Additional requirements include acceptable pressure drop and minimal operating power. Such single-phase, mechanical pumps have been used for circulating heat-transport fluids. Advanced mechanical pumps with similar metrics that are capable of handling two-phase fluids are also desirable. Advanced technologies, such as piezoelectric pumps or electrohydrodynamic (EHD) pumps, offer very high pumping efficiencies (> 90 percent), longevity, and negligible vibration. Reliable check valves would allow the inclusion of a redundant pump, if required by the mission. Improved thermal control valves for regulating the temperature of components on a single-phase loop when the external environment changes are also needed. Miniaturization in this area is also highly desired.

Heat Straps must achieve metrics of approximately 1.4 Watts per Kelvin (W/K) for room-temperature applications to allow improved heat conduction while maintaining mechanical flexibility comparable to existing technology. This technology is applicable to a broad range of applications, but most typically for scientific instruments. Occasionally it is necessary to passively transfer heat from one specific location to another within the spacecraft itself or to spread the heat load across a wider area. In these situations, it is often convenient to use a heat strap, for which the SOA typically uses higher mass materials such as copper and aluminum. Emerging technologies using ultra-high conductivity carbon fibers, artificial diamond films, carbon nanotubes, boron nitride nanotubes, now at technology readiness level (TRL) 2 to 4, or other such materials promise lower mass and thermal conductivity and are largely dependent on materials technology advances are addressed in the technology roadmap for TA 12 Materials, Structures, Mechanical Systems, and Manufacturing.

Heat Switches must achieve at least 10 times improvement in on/off thermal conductivity ratio, for example, goal of 100:1, one fifth of current mass/volume, wider temperature range, and two times higher reliability and lifetime. Various approaches are possible, including mechanical connect/disconnect and gas-gap technologies. Heat switches are often employed with heat straps, or at connection points between equipment. These devices allow, or prevent, a thermally-conductive link. Heat switches may be designed to be purely passive, by opening or closing at specific preselected temperatures, or they may be under active command. Technology improvements are possible and the objectives are typically higher thermal conductivity with lower mass for the device, and higher ratios of conductance in the "on" and "off" conditions.

Heat Pipe Capillary-Based Loops must achieve reliable and efficient heat transfer with multiple heat loads (for example, evaporators) and multiple condensers, such as radiators or heat exchangers in order to be used in a variety of science instruments. A capillary-based, two-phase heat transport technology involves the use of a loop where there are separate liquid and vapor lines and the wick is located only at the evaporator. Such loops are called loop heat pipes (LHP) or capillary pumped loops (CPL). While loops have been built that can transport over 20 kW over a ten-meter length for ground demonstrations, for space applications, almost all two-phase loops have currently been limited to a few hundred watts. These technologies offer significant heat transport over long distances with low temperature drop, are inherently self-regulating, need no mechanical pump, typically have a 100:1 ratio of maximum to minimum heat transport capability, and can last indefinitely. They are capable of very tight thermal control, +/- 0.1°C. All of these advantages are very important for scientific spacecraft that have instruments in need of precise thermal control. Existing LHPs and CPLs on operational spacecraft have only one evaporator and usually only one condenser/radiator. Hence, an important advancement would be the development of LHPs with multiple evaporators and condensers. Current analytical models are functional, predict transients reasonably well, and are useful for design purposes, but more complete models with micro-gravity-validated correlations are desired. This is particularly true for systems with multiple evaporators and condensers. Also, models that can predict temperature instability conditions, such as those associated with certain combinations of evaporator mass, power lift, liquid/vapor front within the condenser, and reservoir design, are desired.


Heat Pipe Capillary Loop

While the vast majority of heat pipe capillary-based loops have been for mid-temperature applications, this technology can also be applied to cryogenic applications. A nitrogen-based cryogenic CPL was flown in October of 1998. This flight was the first in-space demonstration of the cryogenic capillary pumped loops (CCPL), a lightweight heat transport and thermal switching device for future integrated cryogenic bus systems. The device operated successfully between 75 K and 110 K. Future needs include the development of cryogenic heat pipe capillary loops that can operate well with the low surface tension fluids typical of cryogenic applications.

Heat Pumps use energy to transfer heat against a thermal gradient to reject heat to a higher temperature sink. Although ground-based designs that are not reliant on gravity for elements of heat pump operation for example, lubricant management, contaminant control, phase separation, are in use, intermittent operation in microgravity and in severe environments, such as hard vacuum, radiation, and extreme temperatures, are real concerns. Additionally, heat pumping can help reclaim waste heat from equipment at lower temperatures for heating other hardware with higher minimum temperature limits. Exceptionally long life, low mass, high efficiency, and operation with high temperature lifts (50 °C or more) and an increase in coefficient of performance (CoP) for comparable applications of 50 percent are key improvements for space-based heat pump technology.

Vapor Compression Heat Pump for Cascade Distillation

Technical Capability Objectives and Challenges

Once waste heat has been acquired, it must be transported to a heat exchanger for reuse or to a radiator or evaporative cooling system for ultimate rejection to the space environment. The specific technology employed for transport is dependent on the temperature and/or heat flux and thus a wide variety of equipment and techniques can be used. In some cases, heat transfer is not desired or must be tightly controlled and technologies that limit or prevent heat transfer are also critical. Current heat transport systems for human spacecraft generally use a two-loop system, transporting heat from internal systems to an interface heat exchanger, where heat is transferred to an exterior loop and then transported to a heat-storage or heat-rejection system. The requirement for a two-loop system is driven primarily by transport fluid requirements for the internal and external applications. The internal loop must employ a non-toxic fluid to minimize hazards to the crew. The external loop uses fluids that are appropriate for the thermal environments including low freezing temperature and maximized thermal capacity and pumping characteristics. Many robotic spacecraft employ two-phase loops driven by capillary forces, such as heat pipes or loop heat pipes. Single-phase, mechanically pumped systems are also increasingly used. Heat transfer can also be accomplished using conductive materials, such as heat straps and conductive structure, or by using heat pumps to provide thermal lift against a temperature gradient. Insulators are used to prevent heat transfer between spacecraft systems or components and the environment.

The primary challenges in heat transport technologies include transport fluids with performance characteristics that are suitable for human spaceflight missions, reliable fluid pumps for fluid-loop systems, efficient heat pumps to reliably move heat against a thermal gradient, passive conductors and heat pipes that are tailored to specific applications, and high-performance and lightweight insulators that can prevent heat transport between components and between the spacecraft and the space environment. Insulators are covered in several areas of section 14.2, depending on specific needs.

Total Organic Carbon Analyzer Onboard the International Space Station

Benefits of Technology

Developing optimized thermal transport fluids will allow safe, reliable, and reduced-mass transport loop designs for human missions. Advances in heat pump and thermo-electric cooler technologies will allow more efficient and reliable systems that reduce power requirements and allow heat transport against a steeper thermal gradient thus allowing spacecraft to operate in a wider range of environments. Advances in materials will provide more efficient heat switches and heat straps and insulations that will provide more reliable and efficient passive thermal control of spacecraft.

Table 7. 14.2.2 Technology Candidates – not in priority order

TA	Technology Name	Description
14.2.2.1	Heat Transport Fluid	Heat transport fluids that provide optimal thermo-physical properties for pumped loop acquisition, transport, and rejection while providing low toxicity to crew.
14.2.2.2	Advanced Pumps	Long-life pumps for circulating heat transfer fluids.
14.2.2.3	Heat Straps	Mechanical devices with extremely high thermal conductivity.
14.2.2.4	Heat Switches	Remotely-actuated mechanical or gas-based devices with high thermal conductivity on/off ratio.
14.2.2.5	Heat Pipe Capillary Based Loops	Closed, two-phase heat transfer loops serving multiple heat loads and rejecting to multiple thermal sinks, with tight temperature control and minimal temperature drops.
14.2.2.6	Heat Pump	Devices that use energy to transfer heat against a thermal gradient to reject heat to a higher temperature sink.
14.2.2.7	Thermal Electric Coolers (TECs)	Solid-state devices that use the Peltier effect to pump heat against a thermal gradient.
14.2.2.8	In-Situ Thermal Fluids Chemical Analysis	In-situ thermal fluids chemical analysis to monitor thermal transport fluid health status.
14.2.2.9	High-Thermal-Conductivity Thermal Interface Materials	Advanced materials that can provide very high thermal interface conductance and yet also be workable. Vacuum compatible, and not degrade performance through repeated cycling.
14.2.2.10	Micro- and Nano-Scale Heat Transfer Surfaces	Advanced surface treatments to enhance the heat transfer coefficient and/or stability of heat exchangers in contact with a fluid.
14.2.2.11	Integrated Structural, Thermal, and Optical Computer Software	Improved integration of existing software codes, or development of new codes and algorithms, for simulating optical alignment as a function of temperature and materials properties.

TA 14.2.3 Heat Rejection and Energy Storage

This technology area is comprised of nine types of heat rejection and energy storage technologies, including:

Radiator Surface Dust Control must provide a means for greatly reducing the accumulation of in-situ dust on radiators and other sensitive surfaces. The technology includes specialized passive or active control technologies that will reduce or eliminate dust accumulation on a radiator surface. Passive, “Lotus Coating”, or active techniques for cleaning radiator coatings are being pursued; these technologies are currently at TRL 3 to 5 and could be matured to TRL 6 to 8 within a few years. Other technologies enable a number of solutions, such as using waste gas to blow dust off radiators or other vehicle surfaces, electrostatic charging, etc.

Two-Phase Pumped Loop Systems include two-phase heat transport systems for thermal control of large heat loads, such as those required by Rankine cycle power plants. Two-phase thermal control systems are widely used on Earth, but are not widely used in space due to a lack of understanding of the effects of zero gravity.

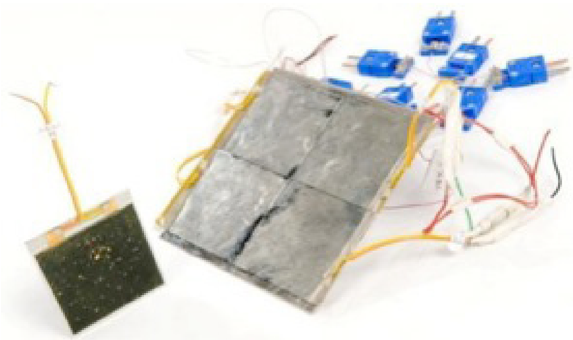
Phase-Change Heat Exchangers include phase-change materials used to store thermal energy during hot phases of cyclic thermal environments for later rejection during cold phases. The primary challenge in developing this technology is managing the phase boundary as the liquid freezes in the heat exchanger to prevent damage due to expansion.

Phase Change Material (PCM) Heat Exchangers

Evaporative Cooling technologies are focused on cooling through a water membrane evaporator. The goal is to provide contamination-insensitive evaporative cooling with low-to-zero consumables to provide a heat sink for space suits and spaceflight vehicles, with 30 percent mass reduction, an order of magnitude increase in contamination load increase tolerance, and 90 percent decrease in consumables. This is required for future long-duration missions with requirements for frequent extravehicular activities (EVAs). These missions will be enabled by significantly reducing consumables mass. The current technology challenge is to incorporate water evaporation through a membrane and reaction of the resulting water vapor in an exothermic sorbent bed. The bed is synchronized to a radiator that rejects heat as the water is adsorbed. The system is then regenerated with heat and gas purge. This entire system must be packaged in a portable life support system volume.

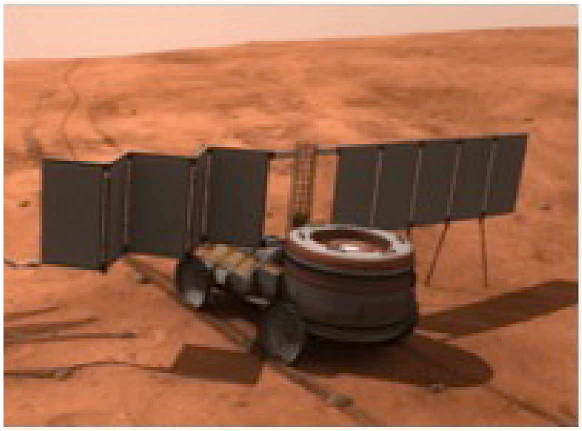
Freezable and Stagnating Radiator (Variable Heat Rejection Radiator Technology) provides technologies with freezable or stagnating and recoverable radiator fluid paths to allow spacecraft radiators to freeze or stagnate during cold mission environments and predictably thaw or restore flow for resumed operation in hot environments. Radiator technologies may be combined with thermal control system architectures and thermal transport fluids to reach the capability goals.

Variable-Geometry Radiators (Variable Heat Rejection Radiator Technology) provide variable heat rejection capability across mission thermal environments and spacecraft heat loads to keep transport fluids from freezing and allow for use of a single-loop spacecraft active thermal control architecture. These technologies are focused on allowing heat rejection turndown by varying the radiating surface's view to the radiative heat sink. Some candidate technologies include micro-louver surfaces, shape-memory alloys, and reliable deployable and retractable systems. The development goal is to provide radiators with a 6:1 (with a stretch goal of 12:1) heat rejection turndown capability, but initial analyses indicate that ratios of 50:1 are feasible. This technology is one of several considered for providing heat rejection turndown capability and is a candidate for demonstration on an International Space Station (ISS) platform in 2018 to advance it to TRL 7. Variable heat rejection is considered an enabling technology in order to reliably restrict and recover heat rejection capability during human and robotic spaceflight missions with wide variation in thermal environments and vehicle heat loads beyond low-Earth orbit (LEO). Radiator technologies may be combined with thermal control system architectures and thermal transport fluids to achieve capability goals.



Electro-Chromic Radiator Test Panel, a Variable Heat Rejection Radiator Technology

Variable Emissivity Radiator (Variable Heat Rejection Radiator Technology) provides variable heat rejection capability across mission thermal environments and spacecraft heat loads, which keep transport fluids or equipment from going below survivable temperatures. For crewed applications, this would allow for use of single-loop spacecraft active thermal control architecture, saving significant mass. For robotic applications, this could prevent loss of electronic equipment due to cold temperatures. Technologies may include variable emissivity coatings or surface treatments that vary optical properties through active voltage commands (electrochromic) or passively with temperature change (thermochromic).



High Temperature Radiator

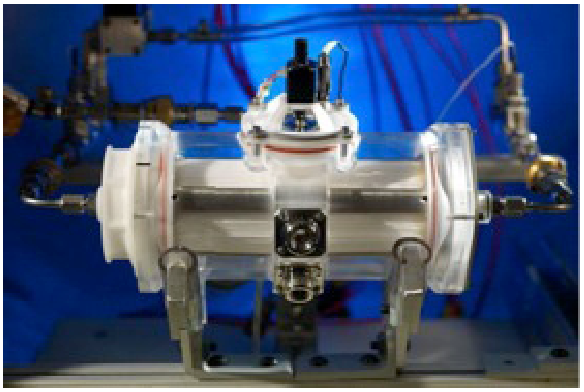
Radiator Repair includes the equipment, materials and processes required to repair spacecraft radiator systems in a space environment in order to maintain heat rejection capability and conserve transport fluids. Current external cooling loop repair technologies are limited to fluid line repair on the ISS. There is currently no radiator panel coolant line repair capability.

80-250° C Variable Conductance Heat Pipe Radiator provides higher-temperature heat rejection typically associated with nuclear power systems.

Specialized thermal control systems will be required for nuclear or other high-power spacecraft. Coordination with TA 2, In-Space Propulsion Technologies, and TA 3, Space Power Energy Storage, will lead to an integrated solution for the thermal control of high-power spacecraft systems. Key thermal components required include high-temperature liquid metal coolants with high thermal conductivity; and large, multi-megawatt deployable radiators. Radiator surfaces will be required to reject heat at high temperatures (~500 K). The SOA does not exist for high-temperature radiator coatings with acceptable longevity and emissivity. This will be critical to a Rankine-cycle thermal nuclear power system. In addition to hardware technologies, analytical models for high-temperature systems are needed.

Technical Capability Objectives and Challenges

Waste heat must be rejected from a spacecraft or surface system to the space environment in order to keep system temperatures within tolerable limits. Heat rejection is accomplished using radiators, evaporators, and sublimators. Thermal energy can also be stored either as latent heat in a phase change material or through sensible heating of a large mass for later use or rejection into a more favorable environment. This can significantly reduce the thermal control system mass by mitigating the effects of peak and minimum thermal loads, as well as the extreme thermal environments. Current heat rejection systems for crewed application may use boilers or sublimators for relatively short-duration mission phases when consumable quantities used by the system trade well against system mass. For most spaceflight missions, radiators are used to reject heat to the space environment. These radiators can be body mounted or deployable, depending on the nature of the mission. Specialty radiators, especially those for high-temperature applications, may take advantage of two-phase technologies. Thermal energy storage devices are used when cyclical thermal environments, like lunar orbit, allow heat to be stored during relatively hot portions of the mission cycle and rejected later when the thermal environments are more suited to rejection.



Space Suit Water Membrane Evaporator

Smart Seal included in a fluid line repair kit for International Space Station on-orbit radiator repair

The primary challenges in radiator heat rejection technologies include the ability to vary radiator heat rejection rates to match mission phase requirements based on heat loads and thermal environments, perform real-time mission repairs of damaged radiator systems, and manage radiator surface optical properties during extended deep-space and surface missions. High-temperature systems pose further challenges for efficient high-temperature radiators and two-phase systems. Challenges for sublimator and evaporator technologies include reduction of contamination sensitivity and recapture of consumables. Thermal energy storage challenges include mass reduction using water as opposed to wax for latent energy storage, and management of freeze and thaw dynamics to assure efficient and reliable performance.

Benefits of Technology

Advances in heat rejection technologies will provide more capable heat rejection systems across a wider range of thermal environments and heat loads. Variable capacity radiators, which may be accomplished through a variety of technologies, offer substantial system-wide benefits. The ability to efficiently store heat for later rejection during some mission phases will allow for optimized design of heat rejection systems. Radiator dust control and repair technologies will help to assure radiator performance throughout the mission. New evaporative cooling technologies will reduce risks resulting from contamination and may lead to consumables recovery, which will result in significant mass savings. High-temperature, two-phase systems and heat pipes will support high-power system requirements, such as nuclear power systems.

Table 8. TA 14.2.3 Technology Candidates – not in priority order

TA	Technology Name	Description
14.2.3.1	Radiator Surface Dust Control	Specialized passive coatings (or active control) that will reduce or eliminate dust on a radiator surface.
14.2.3.2	Two-Phase Pumped Loop System	Two-phase heat transport systems for thermal control of large heat loads, such as those required by Rankine cycle power plants.
14.2.3.3	Phase-Change Heat Exchanger – Phase Change Material Thermal Storage (Heat Sinks & Storage)	Phase-change materials (PCM) used to store thermal energy during hot phases of cyclic thermal environments for later rejection during cold phases.
14.2.3.4	Evaporative Cooling	Evaporative cooling through water membrane evaporator.
14.2.3.5	Freezable/Stagnating Radiator (Variable Heat Rejection Radiator Technology)	Freezable or stagnating and flow-recoverable radiator fluid paths to allow spacecraft radiators to freeze during cold mission environments and predictably thaw for resumed operation in hot environments.
14.2.3.6	Variable-Geometry Radiators (Variable Heat Rejection Radiator Technology)	Variable-geometry radiators allow heat rejection turndown by varying the radiating surface's view to the radiative heat sink.
14.2.3.7	Variable-Emissivity Radiator (Variable Heat Rejection Radiator Technology)	Material coatings or electrical layers that allow control of surface emissivity to manage radiated energy.
14.2.3.8	Radiator Repair	Equipment, materials, and processes required to repair spacecraft radiator systems in a space environment.
14.2.3.9	80-250° C Variable Conductance Heat Pipe Radiator	Heat pipe radiator for high-temperature heat rejection typically associated with nuclear power systems. Passive and variable heat rejection using variable conductance heat pipe technology.
14.2.3.10	Planetary Lander Multi-Phase Thermal Control	Provides survival or extends the life of a spacecraft and components when in an extreme environment and conventional radiative heat rejection is impossible, as the ambient temperature exceeds the spacecraft (e.g. lander) operational temperatures.

TA 14.3: Thermal Protection Systems

Spacecraft require thermal protection systems to survive the extreme flight environments of high-speed ascent through, or entry into, an atmosphere. The challenge of high-speed atmospheric entry requires the use of exotic, engineered materials and structures capable of surviving chemically-reacting high-temperature flows while withstanding the corresponding normal pressure and shear force loading. As entry speeds increase, shock radiation becomes the predominant contributor to entry heating.

TPS solutions for the extreme heating environments (> 5,000 W/cm^2) associated with missions to Venus or the giant planets require constituent materials used in the production of heritage-type TPS. These heritage materials are very limited and production does not currently exist. Technology development is required to enable production of these materials or find suitable replacement materials or development of a new TPS. Currently, there are no mass-efficient solutions for thermal protection systems in the 1,000 to 7,000 W/cm^2 range, corresponding to entry velocities > 11 km/sec. Development of such capability is considered enabling for future beyond LEO missions requiring atmospheric entry. While TPS materials developed for the extreme entry environments associated with heating rates of 5,000 W/cm^2 and greater would provide adequate protection, they are not a mass-efficient solution and will result in a substantial mass penalty (in some cases, prohibitive mass penalty) for a mission using such a TPS. Therefore, considerable technology development is required for mass-efficient TPS to support heating rates from 1,000 to 7,000 W/cm^2 to enable specific future missions, in general, crewed return from beyond LEO, increased mass to Mars, Mars sample return missions, and Venus aerocapture. Deployable heat shield systems represent a relatively new technology that has tremendous potential for enabling specific missions. Development of flexible TPS for operation over a heating range of 50 to > 250 W/cm^2 is required to enable the deployable systems. Beyond high-temperature materials, technologies such as health monitoring systems and in-space repair are necessary to reduce the risk of an often low- or no-fault-tolerant, critical system.

Due to the extreme aerothermochemical-operating environment, advanced analytical tools are required to enable the design of TPS that maintains positive structural margin without levying a burdensome weight to the spacecraft structure. The high-temperature environment typifying TPS performance results in coupled flow, thermal, material, and structural response that cannot be solved separately. Multi-dimensional effects, such as flow, conduction, and recession, play an important role in the performance of many TPS and should also be included in modeling efforts. The development of TPS modeling is a collaborative effort requiring testing. To support model development and validation, tests need to be conducted and the test environments will need to be characterized.

Table 9. Summary of Level 14.3 Sub-Goals, Objectives, Challenges, and Benefits

Level 1		
14.0 Thermal Management Systems	Goals:	Maintain temperatures of a sensor, component, instrument, spacecraft, or space facility within the required limits, regardless of the external environment or the thermal loads imposed from operations.
Level 2		
14.3 Thermal Protection Systems	Sub-Goals:	Protect spacecraft and systems ascent through, or entry into, an atmosphere.

Table 9. Summary of Level 14.3 Sub-Goals, Objectives, Challenges, and Benefits - Continued

Level 3		
14.3.1 Ascent/Entry TPS	Objectives:	Develop mass-efficient TPS for high-speed entry > 11 km/s or for extreme heating environments. Develop flexible and deployable heat shield systems for heat rates from 50 W/cm^2 to 250 W/cm^2.
	Challenges:	Development of a replacement carbon-phenolic material process, and integration of resultant material into a complete vehicle system. Development of fourth generation, environmentally friendly TPS materials and in-space TPS repair technologies. Advanced coatings and/or fibers for the thermal barriers that can operate from 1400° C to 1800° C, and associated fabrication techniques and equipment.
	Benefits:	Provides mass-efficient solutions that meet mission requirements with higher velocity reentry while decreasing associated TPS risks.
14.3.2 Thermal Protection System Modeling and Simulation	Objectives:	Develop advanced analytical tools to enable the design of TPS that maintains positive thermal, recession, and structural margins while minimizing weight to the spacecraft structure.
	Challenges:	Design and analysis of TPS is very demanding because of the highly coupled nature of the system response to an extreme environment. Performing shock radiation modeling of the absorption by the ablation by-products, and turbulence modeling in the presence of ablation by-products. Non-equilibrium chemistry modeling for shock radiation modeling.
	Benefits:	Decreases the mass of TPS by increasing understanding of the reentry environment and its effects on the TPS, leading to a reduced TPS margin requirement.
14.3.3 Thermal Protection System Sensors and Measurement Systems	Objectives:	Develop full-field sensing capabilities that can withstand high temperatures associated with TPS performance without compromising TPS integrity.
	Challenges:	Radiometer operation in an entry heating flux environment that is two to ten times what is currently experienced. Sensor and transmission system capable of operating at temperatures representative of TPS conditions (500° C to 1000° C) with one to five percent resolution. Access to arc jet testing, and the limited performance envelope of ground testing facilities.
	Benefits:	Increases the ability of the TPS to assess health during a mission lifecycle, as well as collect valuable data that is used to reduce modeling and simulation uncertainties, TPS sizing margins, and risks associated with a low-to-no-fault-tolerant, critical spacecraft system.

TA 14.3.1 Ascent/Entry TPS

There are eight types of entry and ascent TPS technologies, including:

Rigid Ablative TPS: This TPS technology focuses on development of mass-efficient TPS capable of performance in the heat flux range of 1,000 to 7,000 W/cm^2 to enable missions such as beyond low Earth orbit (LEO) crew return, higher-mass Mars entry, and dual-pulse heating profiles. Additionally, the ability to manufacture chop-molded carbon-phenolic or a suitable replacement to enable entry environments >7,000 W/cm^2 associated with Venus and giant planets is required.

Rigid Ablative Thermal Protection System

Rigid ablative TPS provides the ability to perform atmospheric entry at high velocities and operating temperatures beyond reusable TPS material limits. Ablative materials provide thermal protection from high-speed atmospheric aerothermal entry heating loads by pyrolysis of in-depth resins and surface ablation to protect the underlying spacecraft structure. Current options include: phenolic impregnated carbon ablator (PICA), Avcoat, SLA-561V, carbon-carbon, and carbon-phenolic, which are limited by factors such as high velocities, dual-pulse heating profiles, high areal mass, manufacturing reliability, and material integration into the system.

SOA for the highest-TRL ablator TPS options are:

SLA-561V: Flown on several Mars entries at heat fluxes of 100 to 200 W/cm^2 and low stagnation pressures (<0.3-atmospheres, or atm).

PICA: Tested up to a heat flux of ~1,400W/cm^2 at low to moderate pressures (approximately 0.3-atm).

Avcoat (reformulated): Has been successfully tested up to 900 W/cm^2 at low-to-moderate stagnation pressures.

Carbon-Carbon: Tested up to 10,000 W/cm^2 for brief exposure.

Carbon-Phenolic: Flown in an environment up to 30,000 W/cm^2 and 7-atm stagnation pressure.

Current material challenges include manufacturing components and integrating them into a system, typified by solutions for gaps between blocks and limitation of block size. PICA is manufactured in discrete blocks of limited size; therefore, for heat shields larger than approximately one meter in diameter, blocks must be individually attached to the heat shield substructure, requiring gaps between blocks to be filled.

Avcoat has high manufacturing labor cost associated with individually hand-filling each honeycomb cell. Other TPS materials, such as SLA-561V, do not have as high heat flux capability and cannot support all missions. For extreme entry conditions associated with robotic missions to Venus and the giant planets, carbon-phenolic heat shields have been used in the past. However, the ability to manufacture a heritage version of the material is limited due to diminished supply of the precursor material and imminent discontinuation of key processes.

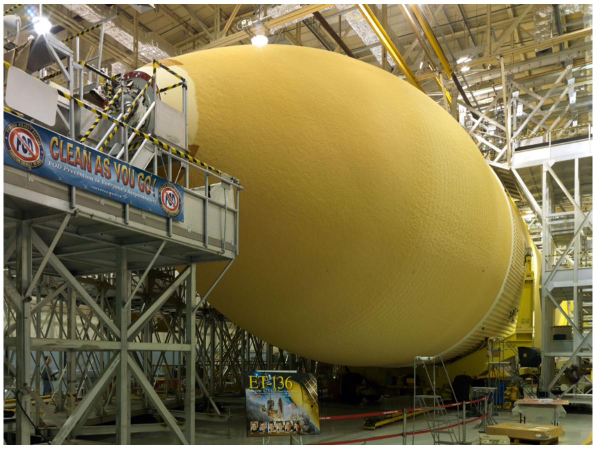
Presently, there is no mass-efficient ablative TPS to support entry environments in the 1,000 to 7,000 W/cm^2 range, which is required to enable crew return beyond LEO, higher-mass delivery to the Martian surface, and dual-pulse heating profiles. Carbon-carbon and carbon-phenolic can perform in this environment but will result in a substantial weight penalty that may make the mission infeasible.

Conformal TPS and woven TPS are two technologies currently under development that can eliminate many of the issues with the SOA rigid ablative TPS while meeting some of the goals stated above. Development of these technologies is encouraged and it is expected that these technologies will be enhancers and enablers for planned missions requiring entry heat loads of 1 kiloWatt per square centimeter (kW/cm^2) to 7 kW/cm^2.

Lastly, technology development is not complete once a material or system achieves high TRL. Development needs to continue to improve sustainability (availability, processing equipment, environmental processing issues) of the SOA materials.

Obsolescence-Driven TPS Materials and Process Development must retain current capability yet provide materials and processes that are free from banned products and meet all standards for constituent materials.

Ascent thermal protection materials, such as Shuttle-era cryo-insulation, primers, and ablators containing now-banned regulated and restricted materials like hydrochlorofluorocarbons (HCFCs) and hexavalent chromium should be replaced while continuing to meet technical performance requirements. Obsolescence-driven TPS materials provide the Agency with the ability to retain current capabilities that will be or are



Obsolescence-Driven Thermal Protection System Materials Development

being lost due to environmental regulation compliance issues. Ongoing development and qualification of third generation, HCFC-based cryo-insulation foams to replace Shuttle-era material systems and their associated processes is required. A proactive search and testing for fourth generation candidates that have no future potential for regulated and restricted status coupled with development of replacement materials free of banned chlorofluorocarbons (CFCs) that meet all standards for constituent materials as well as meeting technical performance requirements, is needed.

Flexible/Deployable TPS is a non-ablative TPS capable of entry environments of 15 to 100 W/cm^2 and an ablative TPS capable of environments of 75 to >250 W/cm^2 to enable higher-mass delivery and higher-altitude surface exploration on Mars and specific Venus robotic missions.

Deployable entry systems can provide an entry drag device unconstrained by launch vehicle shroud size, which reduces the ballistic coefficient and increases the energy dissipation during atmospheric entry, thereby providing increased capabilities in terms of mass delivered, or accessible exploration surface areas. Primary areas of pursuit include:

- Non-ablative (insulative or transpiration-cooled) material concepts with high flexibility and stowability for a heat flux environment up to 100 W/cm^2;
- Ablative material concepts that include high-heat-flux semi-flexibles for a heat flux environment >250 W/cm^2 for mechanically deployed systems and for low-cost application to rigid aeroshells.

In-Space TPS Repair must restore thermal protection capability to a damaged TPS, maintain underlying structural temperatures below limits, and demonstrate a shelf life of one year or longer in order to be used on future crewed missions.

Damage to space vehicle entry thermal protection systems resulting from ascent, on-orbit micrometeoroid and orbital debris (MMOD) exposure, and damage induced through other operations may compromise the ability of the TPS to adequately protect the vehicle and crew during atmospheric entry. Suitable repair technologies are required to restore capability to entry TPS. STA-54 ablator and the tile repair ablator dispenser (TRAD) dispensing system were deployed for potential use as a thermal protection system repair during the Space Shuttle Program but their use for an actual repair was never demonstrated. Similarly, non-oxide adhesive experimental (NOAX), along with application techniques, was developed to repair the orbiter reinforced carbon-carbon (RCC) hot structure. These capabilities were developed for the Space Shuttle Program and neither was maintained nor are they presently available. Even if the previously established repair technology was available today, it is not known whether such repairs are sufficiently robust for return from beyond Earth orbit. Currently, there are no human-rated vehicles in operation in the United States (U.S.) that require thermal protection repair. New in-space TPS repair materials and techniques are required, resulting in a repair sufficient to maintain underlying structural temperature < 350 degrees Fahrenheit (°F) (177°C) for aluminum structures and cyanate ester structures, < 500°F (260°C) for entry heat fluxes > 80 W/cm^2, with a material shelf life greater than one year for near-Earth missions, and longer than the expected mission duration for Mars and asteroid missions.

TPS Integral Health Monitoring System (HMS) technology must provide data from across all critical portions of the system and obtain full-field sensing with negligible weight and volume penalty and minimal intrusion into the TPS while enduring the entry environment. Achieving these goals would enable the technology to be incorporated into the future spacecraft and missions, enabling acquisition of detailed TPS performance data on the host spacecraft and resulting in risk reduction for follow-on missions.

Many spacecraft TPS have flown with sensors; however, the sensors were limited, discrete measurements that constituted another weight and volume penalty for the spacecraft while potentially impacting the TPS functionality. A TPS with an integral HMS capable of providing full-field sensing with negligible weight, volume, and performance penalty would reduce the specific mission risk by providing the ability to assess system health, and also reduce future mission risk by providing data on the system response to the real flight environment during atmospheric entry. The system would require fusion of data from multiple types of sensors for the detection and characterization of any events that could affect TPS functionality and assess the remaining post-event functionality. Sensor density is a valid metric for discrete measurement methods, while resolution would be the corresponding metric for optical field sensing techniques. Wireless sensors and fiber-optic-based sensors are viable candidates to provide the sensing elements of the TPS HMS while incurring negligible volume and mass penalty. Non-intrusive ultrasonic-based sensors also hold promise in providing significant capability to TPS HMS architectures. TPS HMS performance requirements would be specific to the size and function of each spacecraft, with significant metrics being: event detection sensitivity, acceptable event location resolution, as well as resolution requirements for qualitative and quantitative event and damage assessment.

Self-Repairing TPS Materials may serve as part of the TPS for future spacecraft, whether crewed or uncrewed, by increasing the maximum temperature limits of current self-healing materials. Successful development of self-repairing TPS materials would reduce risk in what is currently a critical and low-to-no fault tolerant spacecraft system for every mission requiring atmospheric entry.

One method for reducing risk of damage from MMOD is to design a TPS with self-healing materials that return to their virgin state (or near-virgin state) functionality after damage events. These materials are evident in nature, as several different approaches have been successfully developed and demonstrated in the laboratory environment. The operating temperature limits of these different materials will need to increase in order to be considered viable components of TPS. No current TPS materials are available that incorporate self-healing capabilities. The assessment of the self-healing capability is evaluated by determining mechanical and thermal performance of the material before and after damage events, as well as material continuity, as measured by vacuum retention. At the system level, the effective thermal conductivity and allowable strength would be necessary to characterize for a range of damage events.

Multifunctional TPS must demonstrate mass, volume, or system performance benefits over the uncoupled systems to be used on future spacecraft. Multidisciplinary approaches to traditional systems like TPS hold potential for enabling missions currently out of reach by developing efficient spacecraft structures and systems. There are several functions that may be integrated into the TPS thereby reducing or even eliminating the need for separate systems including structurally integrated TPS, TPS with MMOD or radiation protection, and cryogenic-insulation-coupled TPS (covered in section 14.1). In some instances, TPS may have electrical grounding requirements. See TA 9.1.1.4, TA 9.1.2.3, TA 12.1.4.3, and TA 10.1.3.5 for more information.

As an example, consider the potential offered by a structurally integrated TPS. Developing a TPS wherein components are able to carry a significant fraction of the structural loading in addition to the aerothermal loads could significantly decrease the structural mass fraction of a spacecraft and allow increased payload. Preliminary work has been performed on multiple concepts, like composite-overwrapped insulation and sandwich panels utilizing various core configurations of honeycomb or trusses, but system trade studies and concept developments are needed to mature the technology. Manufacturing techniques, including the ability to construct compound curved surfaces, would need development, as would panel or component joining methods that would allow load transfer. Scalability is always a concern with structural concepts, and structurally integrated TPS is no exception. Non-destructive evaluation (NDE) for a structurally-integrated TPS would be important because structurally-integrated TPS would likely have a wide variety of failure modes resulting from constituent material dissimilarities and load transfer between and within components. Both composite and metallic hot structures that carry aerodynamic loading and protect the vehicle from heating should be considered. See TA 10.1.1.5, TA 12.1.1.2, and TA 12.2.5 for more information.

High-Temperature Seals and Thermal Barriers must demonstrate higher operating temperatures, improved resiliency, wear resistance, and durability resulting in increased damage tolerance at higher temperatures to enable TPS that can survive the high-velocity atmospheric entry environment of crewed return from beyond LEO, or uncrewed probes to other planets.

High-temperature seals and thermal barriers are an important component of TPS on many spacecraft and are required to inhibit conductive and convective heat transfer through interfaces in a TPS or between engine components to protect underlying temperature-sensitive structures. TPS materials on the Space Shuttle Orbiter were thicker, which allowed thermal barriers to be larger and/or installed further inboard from the outer mold line, and therefore subject to lower temperatures. The TPS concepts for current DRMs involve thinner, ablative heat shields that are subject to higher entry velocities and, therefore, harsher operating conditions. Current higher-temperature thermal seals and barriers utilize integral polycrystalline knitted metallic wire spring tubes and ceramic fabrics, and have operating limits of 1400° C for short duration and 900° C for one to three flight cycle reusability before wear necessitates replacement. For single-use thermal barriers and seals, operating temperatures need to increase to 1800° C.

Technical Capability Objectives and Challenges

Ascent/entry TPS includes the materials, structures, and systems used to survive high-speed atmospheric flight. There are multiple approaches to protecting a spacecraft from high temperature flow: ablation, insulation, high-temperature structural materials, and aerodynamic decelerators that reduce the heating by decreasing the ballistic coefficient. While many TPS architectures have flown successfully, some currently in use may not scale for other missions.

Heat Shield

Ascent TPS solutions are currently adequate to address the ascent heating environment, with the exception of the availability of specific constituent materials required for the production of lightweight ascent TPS that are being phased-out because of environmental concerns. Technology development will be required to find adequate replacements for these obsolete materials.

Adequate TPS solutions are also available for the aftbody region of blunt-cone type aeroshells; therefore, no technology development is required in this area. Ablative materials exist for entry velocities in the range of 11 km/sec and lower for the forebody heat shield of a blunt-cone type aeroshell; however, additional development will improve the manufacture, installation, and performance of these materials.

Benefits of Technology

Spacecraft require thermal protection systems to survive the extreme flight environments of high-speed ascent through, or entry into, an atmosphere. New thermal protection materials and systems are needed to provide mass-efficient solutions that meet mission requirements with higher velocity reentry while decreasing TPS associated risks.

Table 10. TA 14.3.1 Technology Candidates – not in priority order

TA	Technology Name	Description
14.3.1.1	Rigid Ablative Thermal Protection System	Materials provide thermal protection from high-speed atmospheric aerothermal entry heating loads by pyrolysis of in-depth resins and surface ablation to protect the underlying spacecraft structure.
14.3.1.2	Obsolescence-Driven Thermal Protection System Materials	Ascent thermal protection materials, such as Shuttle-era cryo-insulation, primers and ablators containing now banned, regulated, or restricted materials such as hydrochloroflourocarbons (HCFCs) and hexavalent chromium, require replacement while continuing to meet technical performance requirements.
14.3.1.3	Flexible/Deployable Thermal Protection System	Flexible heat shields for deployable systems provide higher heat flux capability to support a wider range of missions.

Table 10. TA 14.3.1 Technology Candidates – not in priority order - Continued

TA	Technology Name	Description
14.3.1.4	In-Space Thermal Protection System Repair	Damage to space vehicle entry TPS resulting from ascent, on-orbit micrometeoroid and orbital debris exposure, and damage induced through other operation may compromise the ability of the TPS to adequately protect the vehicle and crew during atmospheric entry. Suitable repair technologies are required to restore capability to entry TPS.
14.3.1.5	Thermal Protection System Integral Health Monitoring System	A TPS with an integral health monitoring system (HMS) would reduce mission risk by automatic impact detection, localization, and evaluation.
14.3.1.6	Self-Repairing Thermal Protection System Materials	TPS that can return to virgin state without external intervention using self-healing materials.
14.3.1.7	Multifunctional Thermal Protection Systems	A TPS that provides increased mass efficiency by incorporating other functions, like structural load-carrying capacity or cryogenic insulation.
14.3.1.8	High Temperature Seals and Thermal Barriers	TPS with higher operating temperatures, improved resiliency, wear resistance, and durability resulting in increased damage tolerance at higher temperatures.

TA 14.3.2 Thermal Protection System Modeling and Simulation

Coupled Multi-Dimensional TPS Modeling analysis techniques for flow, material response, thermal, and structural analysis of TPS performance in flight and ground test environments will reduce analysis time and cost and improve the fidelity of the analysis, resulting in savings of TPS mass by reducing analysis uncertainty.

Current analysis methodologies use independent codes for each discipline, requiring data to be transferred from code to code, resulting in both mesh transfer and fidelity issues. Most material response models are operated as one-dimensional models and transferred to three dimensional (3D) thermal response codes. Some multi-dimensional material response codes have been developed but are not fully tested or routinely used for mission support and analysis. Multi-dimensional, multi-physics analysis tools for coupled aerothermodynamic, material response, thermal, and structural solutions for TPS components are required to improve analysis fidelity, resulting in smaller TPS design margins, lower-mass TPS, and reduced risk.

Shock Radiation Modeling analysis tool advancement is required to reduce prohibitively large uncertainties associated with shock radiation during high-velocity atmospheric entry. Advances in physics-based or empirical modeling of TPS heating due to shock radiance during high-velocity entry is needed to reduce TPS sizing margins associated with shock radiance uncertainty. The magnitude of the heat flux resulting from high-speed entry shock radiance can be understood as being approximately proportional to the velocity to the eighth power, with the uncertainty approximately proportional to the velocity to the ninth power. It is therefore easy to understand that the radiant heat flux can dominate the convective heat flux on the forebody, but it can also exceed 100 percent of the convective heat flux on the aft body. The uncertainty is very dependent on the flight case. For high-speed Earth return, the uncertainty is approximately 35 percent. Mars return is closer to +80 percent/-50 percent. Any flight program would need significant investment in quantifying the uncertainty of radiation simulations. Several sources contribute to the uncertainty. Radiant flux absorption from the ablation products is one of the key drivers and represents approximately half of the uncertainty. The rest of the uncertainty is due largely to turbulence modeling and precursor absorption that is not being modeled in the flow field, but would occur depending on the participating medium composition.

Technical Capability Objectives and Challenges

Due to the extreme aerothermochemical operating environment, advanced analytical tools are required to enable the design of TPS that maintains positive thermal, recession, and structural margins but without levying a burdensome weight to the spacecraft structure. The high-temperature environment typifying TPS performance results in coupled thermal, material, flow, and structural response of the TPS. Multi-dimensional effects, such as conduction, recession, and flow play an important role in the performance of many TPS and must also be included in modeling efforts. TPS design and analysis in the future will increasingly rely on modeling because of inherent limitations of ground testing facilities.

The development of TPS modeling is a collaborative effort requiring testing. To support the development and validation of models, tests will need to be conducted and the test environments will need to be characterized better than have been done in the past. Additionally, thermal test facilities may need to be improved and expanded based on mission requirements. Any plan for TPS modeling development should include test capability characterization and development.

The design and analysis of TPS is very demanding because of the highly-coupled nature of the system response to an extreme environment that is difficult, in some cases prohibitive, and costly to completely replicate in ground test facilities, and expensive to replicate in representative flight tests. Advances are needed in TPS design and analysis tools to reduce or eliminate prohibitive sizing margins.

Benefits of Technology

TPS performance involves complex, multi-physics loading and response in extreme environments that is not fully characterized. Modeling and simulation tools are required to decrease the mass of TPS by increasing our understanding of the reentry environment and its effects on these systems, leading to a reduced TPS margin.

Table 11. TA 14.3.2 Technology Candidates – not in priority order

TA	Technology Name	Description
14.3.2.1	Coupled Multi-Dimensional Flow/ Material Response/Thermal/ Structural Analysis	Improved analysis techniques for TPS performance in flight and ground test environments will reduce analysis time and cost and improve the fidelity of the analysis, resulting in savings of TPS mass by reducing uncertainty.
14.3.2.2	Shock Radiation Modeling	Analytical modeling of entry shock radiance.

TA 14.3.3 Thermal Protection System Sensors and Measurement Systems

Technical Capability Objectives and Challenges

The high-speed reentry environment from beyond LEO cannot be fully reproduced in ground test facilities. Advanced sensors capable of operating in this harsh environment are necessary for flight tests as well as missions to provide the data necessary for model validation.

Sensors and measurement systems are needed for TPS development and in-space TPS health assessment. Current design challenges drive sensor development and system architecture advancement to provide high-fidelity TPS performance data in flight environments at or beyond the limits of previous flight experience. The resulting reentry environments include effects that are not as well understood and require not just improved analytical tools, but instrumentation that can provide test and flight data to validate the model against the real-world environment, and subsequently reduced sizing margins due to large uncertainties.

Radiometers/Spectrometers must measure entry shock radiance and spectral energy content during exo-LEO atmospheric entry. Radiometers must function properly in entry environment heating flux two to ten times that currently experienced with an integrated radiance error < +/-20 percent.

Entry shock-layer radiation is a function of vehicle size and reentry velocity. The larger the vehicle, the higher the velocity and, consequently, the higher the heating. Measurements may also be useful on back shell TPS for Mars entry missions. Mars return missions, depending on the mission architecture, result in higher return velocities and drive the need for these measurements. Uncertainties with shock-layer radiation grow exponentially to the ninth power with increased entry velocities. At these high reentry velocities the shock-layer radiation dominates heat shield heating. Entry shock radiance measurement capability is needed to provide the understanding necessary to design TPS with smaller margins (i.e., lower mass), and reduced risks associated with TPS for high-velocity atmospheric entry. Specifically, measurement of entry shock radiance and spectral energy content during exo-LEO atmospheric entry is required.

Wireless High-Temperature Sensors must increase the operating temperature limit, both for sensing and transmitting, of the sensor and any associated power, energy harvesting, or communication components to above 500°C to provide in-situ, full-field sensing for most DRMs and TPS systems of interest.

Spacecraft that are equipped with full-field TPS sensing solutions would have increased flight safety with the increased ability to assess TPS health, and potentially inform in-space repair or modified operation to compensate for damage. This would benefit future spacecraft with detailed TPS performance data that would enable more optimized solutions based on the data itself or design tools that are validated with more relevant flight data. Wireless high-temperature sensors could provide full-field sensing of a variety of parameters including, temperature, pressure, strain, and heat flux, for TPS with minimal mass and volume penalty. Room-temperature wireless sensing systems are commercially available. Technology advancements have brought the state of the art to laboratory demonstrations of wireless radio frequency identification (RFID) sensors at 900°C, with 500°C circuitry based on silicon carbide electronics using external power at 475°C for 22 days, and a wireless pressure sensor with limited power scavenging for one hour at 475°C. The metrics that will need to be considered in high-temperature wireless sensor development for TPS applications will be the maximum number of sensors per area that can be accommodated by a given spacecraft TPS without compromising performance (if embedded), the maximum operating temperature, minimum sensor and system weight achievable for the operating limits, and the signal integrity possible when transmitting over the required distance for full dataset receipt. The challenge is to develop a sensor and transmission system capable of operating at temperatures representative of TPS conditions (500-1000°C) with one to five percent resolution. This would also require high-temperature packaging, electronics, resonator materials, antennas, communication approaches, and development of integration options into a particular spacecraft TPS.

Fiber Optic High-Temperature Sensors must increase the temperature limits of the sensing elements as well as the host fiber to provide full-field sensing in the temperature regimes and locations of interest in most spacecraft TPS for current DRMs.

Full-field sensing of TPS has not been possible to date, despite the clear benefits, due to the prohibitive weight and volume penalty associated with traditional systems. However, fiber-optic-based sensors provide full-field sensing at negligible weight and volume penalty due to the sensing elements multiplexed on the fibers. These systems have been flown on aircraft, but have not flown in space, and are not currently available with operating temperature limits that allow utilization in TPS environments (300°C sensor maximum temperature, 40°C maximum system temperature). Recent developments have demonstrated thermally stable sensing elements capable of measurements at 1000° C in standard silica fibers, and at 2000° C in sapphire fibers. Advancing these newly-developed sensing elements to provide the full-field sensing capabilities within operating temperature limits representative of TPS performance would provide a sensing system that could provide full-field TPS data with negligible mass, volume, and power consumption penalty, while also being inherently immune to electromagnetic interference (EMI).

Non-Intrusive Recession and Temperature Sensors will provide the ability to refine TPS recession and temperature measurements to enhance designs for high-velocity entry, resulting in lower margins and mass for all missions requiring an ablative TPS.

Recession and temperature sensors measure the amount of recession, recession rate, and temperature at a specific location in the TPS. Current technology uses intrusive instrumentation inserted into the TPS, which can have an impact on the actual recession rate, amount of recession, and temperature. In addition, insertion of the instrumentation into the TPS requires alterations to the heat shield TPS, which can raise concerns about the integrity of the TPS at the instrumentation installation site. A non-intrusive instrumentation system will alleviate concerns about the instrumentation influencing the measurement and concerns about the integrity of the TPS. Non-intrusive measurement techniques are under development using ultrasonic techniques and have been demonstrated in laboratory and limited ground test facilities. The non-intrusive system must function in a flight environment with a flight-type heat shield structure without unacceptably increasing risk to the TPS. The system must also operate within flight volume, power, and mass requirements. No current flight capability for a non-intrusive system has been developed, and only limited ground testing has been performed with laboratory scale systems; however, the results are promising and future development is recommended. In addition to the challenges of size, weight, and power, additional challenges for the development of a non-intrusive system are the cost and access to arc jet testing, and the limited performance envelope of the ground test facilities.

Benefits of Technology

The benefits to future missions provided by the sensors and measurement system technologies increase the ability of the TPS to assess health during a mission lifecycle, as well as collect valuable data that can be used to reduce modeling and simulation uncertainties, TPS sizing margins, and risks associated with a low-to-no-fault-tolerant, critical spacecraft system.

Table 12. TA 14.3.3 Technology Candidates – not in priority order

TA	Technology Name	Description
14.3.3.1	Radiometers/Spectrometers	Measure shock-layer radiation. Shock-layer radiation is a function of vehicle size and reentry velocity. The larger the vehicle, the higher the velocity and, consequently, the higher the heating. Measurements may also be useful on backshell TPS for Mars entry missions. Mars return missions, depending on the mission architecture, result in higher return velocities and therefore drive the need for these measurements.
14.3.3.2	High-Temperature Sensors – Wireless	Full-field sensing of a variety of parameters including, temperature, pressure, strain, and heat flux, for TPS with minimal mass and volume penalty.
14.3.3.3	High-Temperature Sensors – Fiber Optic	Full-field temperature sensing using highly-multiplexed fiber Bragg gratings on optical fibers.
14.3.3.4	Non-Intrusive Recession and Temperature Sensors	Recession sensors measure the amount of recession and recession rate at a specific location in the TPS. Develop technology for non-intrusive measurement of both recession and temperature.

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Appendix

Acronyms

3D	Three-Dimensional
ADR	Adiabatic Demagnetization Refrigeration
BAC	Broad Area Cooling
CCPL	Cryogenic Capillary Pumped Loops
CFC	ChloroFluoroCarbons
CoP	Coefficient of Performance
CPL	Capillary Pumped Loops
DRM	Design Reference Mission
ECLSS	Environmental Control and Life Support Systems
EHD	ElectroHydroDynamic
EFT-1	Exploration Flight Test 1
EMI	ElectroMagnetic Interference
EMU	ExtraVehicular Mobility Unit
EVA	ExtraVehicular Activity
FBCE	Flow Boiling and Condensation Experiment
GEO	GEOsynchronous orbit
HCFC	HydroChloroFluorocarbon
HMS	Health Monitoring System
IR	InfraRed
ISRU	In-Situ Resource Utilization
ISS	International Space Station
LEO	Low-Earth Orbit
LHP	Loop Heat Pipes
MLI	Multi-Layer Insulation
MMOD	Micrometeoroid and Orbital Debris
NDE	Non-Destructive Evaluation
NOAX	Non-Oxide Adhesive Experimental
OCT	Office of the Chief Technologist
PCM	Phase Change Material
PICA	Phenolic Impregnated Carbon Ablator Reinforced
RCC	Carbon-Carbon
RFID	Radio Frequency IDentification
RPCB	Reduced Pressure Cryogen Bath
SBIR	Small Business Innovative Research
SOA	State Of the Art
STOP	Structural, Thermal, and OPtical
SWME	Space Suit Water Membrane Evaporator
TA	Technology Area

TABS	Technology Area Breakdown Structure
TCS	Thermodynamic Cryogen Subcooler
TEC	ThermoElectric Cooler
TPS	Thermal Protection Systems
TRAD	Tile Repair Ablator Dispenser
TRL	Technology Readiness Level
U.S.	United States

Abbreviations and Units

Abbreviation	Definition
°	Degrees
ε*	Effective Emissivity
atm	Atmospheres
C	Celsius
cm^2	Square Centimeters
CO_2	Carbon Dioxide
F	Fahrenheit
G	Gravity
g	Grams
H_2	Hydrogen
Hr	Hour
I_{sp}	Specific Impulse
K	Kelvin
kg	Kilograms
kJ	KiloJoules
kJ/kg	KiloJoules per Kilogram
km/sec	Kilometer per Second
kW	KiloWatt
kW/cm^2	KiloWatt per Square Centimeter
LH_2	Liquid Hydrogen
m	Meters
m^2	Square Meters
mK	MilliKelvin
mm	Millimeter
mW	MilliWatt
psia	Pounds per Square Inch Absolute
W	Watts
W/cm^2	Watts per Square Centimeter
W/K	Watts per Kelvin
W/m	Watt per Meter
wt	Weight
W/W	Watts per Watt

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Technology Candidate Snapshots

14.1 Cryogenic Systems
14.1.1 Passive Thermal Control

14.1.1.1 Load Responsive Insulation

TECHNOLOGY

Technology Description: Multi-environment thermal insulation for spacecraft cryogenic propellant tanks.

Technology Challenge: Lightweight vacuum shell design creates challenges to vacuum leakage prevention around penetrations and tank supports.

Technology State of the Art: Phase 2 Small Business Innovative Research project completed.

Parameter, Value:

Heat loss per unit area in one atmosphere: 29.3W/m^2.

Heat loss per unit area in vacuum: 4.8 W/m^2 (without outer multilayer insulation).

Combined mass per unit area: 1.2 kg/m^2.

TRL 5

Technology Performance Goal: Increase insulation while reducing mass.

Parameter, Value:

Heat loss per unit area less than 50 W/m^2 in one atmosphere and less than 0.2 W/m^2 in vacuum.

TRL 6

Technology Development Dependent Upon Basic Research or Other Technology Candidate: None

CAPABILITY

Needed Capability: Single cryogenic insulation for ground and flight.

Capability Description: Provide a single insulation solution that is responsive to thermal loads from launch (one atmosphere), ascent and on orbit (vacuum) which reduces launch mass.

Capability State of the Art: Currently use foam insulation for one atmosphere environment and multi-layer insulation for the vacuum environment.

Parameter, Value:

Heat loss per unit area in one atmosphere: 63 W/m^2.

Heat loss per unit area in vacuum: 0.21 W/m^2 (with 45 layers of multi-layer insulation), and combined mass per unit area: 2.1 kg/m^2.

Capability Performance Goal: Heat loss per unit area less than current foam in a one atmosphere environment and less than traditional multilayer insulation in a vacuum environment without increase in mass.

Parameter, Value:

On-pad heat loss per unit area of less than 30 W/m^2 and on-orbit heat loss of less than 1-1.5 W/m^2, and a mass per unit area of less than 2.5 kg/m^2.

Technology Needed for the Following NASA Mission Class and Design Reference Mission	Enabling or Enhancing	Mission Class Date	Launch Date	Technology Need Date	Minimum Time to Mature Technology
Exploring Other Worlds: DRM 6 Crewed to NEA	Enhancing	2027	2027	2021	3 years
Exploring Other Worlds: DRM 7 Crewed to Lunar Surface	Enhancing	2027	2027	2021	3 years
Exploring Other Worlds: DRM 8 Crewed to Mars Moons	Enhancing	2027	2027	2021	3 years
Planetary Exploration: DRM 8a Crewed Mars Orbital	Enhancing	2033	--	2027	3 years
Planetary Exploration: DRM 9 Crewed Mars Surface Mission (DRA 5.0)	Enhancing	2033	--	2027	3 years
Planetary Exploration: DRM 9a Crewed Mars Surface Mission (Minimal)	Enhancing	2033	--	2027	3 years

14.1 Cryogenic Systems
14.1.1 Passive Thermal Control

14.1.1.2 Wrapped Insulation

TECHNOLOGY

Technology Description: Thermal insulation for cryogenic tubing.

Technology Challenge: Developing high-performance insulation that can be used for cryogen plumbing and components in space.

Technology State of the Art: Phase II Small Business Innovative Research project in progress.

Parameter, Value:
0.37 W/m less heat flux than state of the art insulation.

TRL
3

Technology Performance Goal: Reduced heat loss rate.

Parameter, Value:
Heat loss per unit length less than 0.25 W/m.

TRL
6

Technology Development Dependent Upon Basic Research or Other Technology Candidate: None

CAPABILITY

Needed Capability: High-performance thermal insulation for cryogenic piping.

Capability Description: Provide high-performance insulation that can be wrapped around pipes and plumbing components for space environment.

Capability State of the Art: Current piping is typically wrapped with multi-layer insulation.

Parameter, Value:
Heat leak per unit length: 1 W/m.

Capability Performance Goal: Reduce the heat loss per unit length of tubing and small-diameter components.

Parameter, Value:
Heat leak per unit length less than 0.25 W/m.

Technology Needed for the Following NASA Mission Class and Design Reference Mission	Enabling or Enhancing	Mission Class Date	Launch Date	Technology Need Date	Minimum Time to Mature Technology
Into the Solar System: DRM 5 Asteroid Redirect – Crewed in DRO	Enhancing	2022	2022	2015-2021	5 years
Exploring Other Worlds: DRM 6 Crewed to NEA	Enhancing	2027	2027	2021	5 years
Exploring Other Worlds: DRM 7 Crewed to Lunar Surface	Enhancing	2027	2027	2021	5 years
Exploring Other Worlds: DRM 8 Crewed to Mars Moons	Enhancing	2027	2027	2021	5 years
Planetary Exploration: DRM 8a Crewed Mars Orbital	Enhancing	2033	--	2027	5 years
Planetary Exploration: DRM 9 Crewed Mars Surface Mission (DRA 5.0)	Enhancing	2033	--	2027	5 years
Planetary Exploration: DRM 9a Crewed Mars Surface Mission (Minimal)	Enhancing	2033	--	2027	5 years

14.1 Cryogenic Systems
14.1.1 Passive Thermal Control

14.1.1.3 Insulation with Micrometeoroid and Orbital Debris Protection

TECHNOLOGY

Technology Description: Spacecraft cryogenic propellant-tank thermal insulation with micrometeoroid and orbital debris protection (MMOD).

Technology Challenge: Developing multi-purpose insulation that can be used for cryogen propellant tanks.

Technology State of the Art: Phase I Small Business Innvoative Research project completed.

Parameter, Value: Mass per unit area: 24% of SOA shield and insulation with equivalent MMOD stopping power and equivalent heat loss per unit area.

TRL 3

Technology Performance Goal: Reduced heat loss with MMOD Protection.

Parameter, Value:

On-orbit heat loss of less than 1-1.5 W/m^2 and a Probability of non-penetration of greater than that determined by mission.

TRL 6

Technology Development Dependent Upon Basic Research or Other Technology Candidate: None

CAPABILITY

Needed Capability: Thermal insulation integrated with hypervelocity impact shielding.

Capability Description: Integrated design that provides both thermal insulation and MMOD impact protection.

Capability State of the Art: Currently, separate MMOD shields are required.

Parameter, Value:

Conventional multi-layer insulation heat loss per unit area: 0.1W/m^2;
Multi-shock shield stopping power: 0.54 cm diameter;

Particle at 7 km/sec;

Combined mass per unit area: 33 kg/m^2

Capability Performance Goal: Heat leak per unit area less than conventional multi-layer insulation and adequate hypervelocity impact protection to ensure mission success.

Parameter, Value:

On-orbit heat loss of less than 1-1.5 W/m^2 and a Probability of non-penetration of greater than that determined by mission.

Technology Needed for the Following NASA Mission Class and Design Reference Mission	Enabling or Enhancing	Mission Class Date	Launch Date	Technology Need Date	Minimum Time to Mature Technology
Into the Solar System: DRM 5 Asteroid Redirect – Crewed in DRO	Enhancing	2022	2022	2015-2021	4 years
Exploring Other Worlds: DRM 6 Crewed to NEA	Enhancing	2027	2027	2021	4 years
Exploring Other Worlds: DRM 7 Crewed to Lunar Surface	Enhancing	2027	2027	2021	4 years
Exploring Other Worlds: DRM 8 Crewed to Mars Moons	Enhancing	2027	2027	2021	4 years
Planetary Exploration: DRM 8a Crewed Mars Orbital	Enhancing	2033	--	2027	4 years
Planetary Exploration: DRM 9 Crewed Mars Surface Mission (DRA 5.0)	Enhancing	2033	--	2027	4 years
Planetary Exploration: DRM 9a Crewed Mars Surface Mission (Minimal)	Enhancing	2033	--	2027	4 years

14.1 Cryogenic Systems
14.1.1 Passive Thermal Control

14.1.1.4 Cooled Insulation for Reduced & Zero Boil Off

TECHNOLOGY

Technology Description: Insulation integrated with broad area cooling (BAC) shields and/or hydrogen (H_2) vapor cooled shields.

Technology Challenge: Flight demonstrated insulation.

Technology State of the Art: Ground testing has demonstrated zero boil-off (ZBO) with a liquid oxygen tank and reduced boil-off (RBO) with a liquid hydrogen tank.

Parameter, Value:

Liquid RBO heat loss: 0.8 and 0.65 W/m^2 with conventional multi-layer insulation (MLI) and advanced MLI, respectively.

Liquid oxygen ZBO: 0 W/m^2.

Liquid hydrogen ZBO: to be demonstrated.

TRL: 5

Technology Performance Goal: Reduce or eliminate propellant loss.

Parameter, Value:

RBO reduction = 66%

ZBO reduction = 100%

TRL: 6

Technology Development Dependent Upon Basic Research or Other Technology Candidate: None

CAPABILITY

Needed Capability: Cryogenic insulation integrated with BAC shields.

Capability Description: Provide large-scale insulation that can be integrated with BAC and/or H_2 vapor cooled shields for large diameter tanks.

Capability State of the Art: Cyrogenic propellant tanks are currently insulated with multi-layer insulation.

Parameter, Value:

Liquid hydrogen tank heat loss per unit area: 0.21 W/m^2.

Capability Performance Goal: RBO applications are to reduce propellant loss and ZBO applications are to eliminate boil-off.

Parameter, Value:

RBO reduction = 66%

ZBO reduction = 100%

Technology Needed for the Following NASA Mission Class and Design Reference Mission	Enabling or Enhancing	Mission Class Date	Launch Date	Technology Need Date	Minimum Time to Mature Technology
Exploring Other Worlds: DRM 6 Crewed to NEA	Enabling	2027	2027	2021	2 years
Exploring Other Worlds: DRM 7 Crewed to Lunar Surface	Enabling	2027	2027	2021	2 years
Exploring Other Worlds: DRM 8 Crewed to Mars Moons	Enabling	2027	2027	2021	2 years
Planetary Exploration: DRM 8a Crewed Mars Orbital	Enabling	2033	--	2027	2 years
Planetary Exploration: DRM 9 Crewed Mars Surface Mission (DRA 5.0)	Enabling	2033	--	2027	2 years
Planetary Exploration: DRM 9a Crewed Mars Surface Mission (Minimal)	Enabling	2033	--	2027	2 years

14.1 Cryogenic Systems
14.1.1 Passive Thermal Control

14.1.1.5 Modeling for Multi-Layer Insulation

TECHNOLOGY

Technology Description: Empirical equations for low-temperature multi-layer insulation (MLI).

Technology Challenge: Challenges include demonstrating a model below 100 K conditions.

Technology State of the Art: Equipment to perform these tests has been built but needs final assembly and calibration.

Parameter, Value:
Accuracy of heat loss per unit area: actual is 2.5 to 5 times lower than prediction.

TRL 4

Technology Performance Goal: Develop empirical equations that accurately predict multilayer insulation performance in the 20 K to 90 K range.

Parameter, Value:
Accuracy equivalent to existing equations for the 90 K to 300 K range.

TRL 6

Technology Development Dependent Upon Basic Research or Other Technology Candidate: None

CAPABILITY

Needed Capability: Equations that are accurate of multi-layer insulation heat loss in the 20 K to 90 K range.

Capability Description: Provide prediction of MLI performance in the 20 K to 90 K range.

Capability State of the Art: MLI heat loss is typically predicted using empirical equations developed in 1974.

Parameter, Value:
Accuracy of heat loss per unit area: (actual is 2.5 to 5 times lower than prediction).

Capability Performance Goal: Develop empirical equations that accurately predict multilayer insulation performance in the 20 K to 90 K range.

Parameter, Value:
Accuracy equivalent to existing equations for the 90 K to 300 K range.

Technology Needed for the Following NASA Mission Class and Design Reference Mission	Enabling or Enhancing	Mission Class Date	Launch Date	Technology Need Date	Minimum Time to Mature Technology
Into the Solar System: DRM 5 Asteroid Redirect – Crewed in DRO	Enhancing	2022	2022	2015 - 2021	2 years
Exploring Other Worlds: DRM 8 Crewed to Mars Moons	Enhancing	2027	2027	2021	2 years
Planetary Exploration: DRM 8a Crewed Mars Orbital	Enhancing	2033	--	2027	2 years

14.1 Cryogenic Systems
14.1.1 Passive Thermal Control

14.1.1.6 Low Thermal Conductivity Structural Supports

TECHNOLOGY

Technology Description: Structural supports for cryogenic propellant tanks that have low heat loss.

Technology Challenge: Challenges include eliminating end fitting debonding.

Technology State of the Art: Small-scale struts constructed with S2-glass/8552 tubes bonded to Titanium6-4 end fittings have been ground tested. Some end fitting debonding occurred.

Parameter, Value:
Minimize heat loss but still have adequate strength and stiffness for flight.

TRL 4

Technology Performance Goal: Minimize heat loss but still have adequate strength and stiffness for flight.

Parameter, Value:
Conductivity near zero at 20 K.

TRL 6

Technology Development Dependent Upon Basic Research or Other Technology Candidate: None

CAPABILITY

Needed Capability: High structural strength and stiffness with low thermal conductivity at 20 K.

Capability Description: Provide high-strength structural support for cryogenic devices that are capable of supporting launch loads while minimizing heat loss due to thermal conductivity.

Capability State of the Art: Currently cryogenic propellant tank supports have not been designed to minimize heat loss for long-duration flights.

Parameter, Value:
Minimal thermal conductivity with adequate strength and stiffness for flight.

Capability Performance Goal: Struts and/or skirts with low thermal conductivity capable of supporting large cryogenic propellant tanks ~8m diameter, that minimize the propellant boil-off rate.

Parameter, Value:
Integrated system, which includes advanced tank insulation, propellant mass boil-off rate: Passive loss rate less than 4% mass/month, with active cooling loss rate less than 1.5% mass/month.

Technology Needed for the Following NASA Mission Class and Design Reference Mission	Enabling or Enhancing	Mission Class Date	Launch Date	Technology Need Date	Minimum Time to Mature Technology
Exploring Other Worlds: DRM 8 Crewed to Mars Moons	Enhancing	2027	2027	2021	2 years
Planetary Exploration: DRM 8a Crewed Mars Orbital	Enhancing	2033	--	2027	2 years
Planetary Exploration: DRM 9 Crewed Mars Surface Mission (DRA 5.0)	Enhancing	2033	--	2027	2 years
Planetary Exploration: DRM 9a Crewed Mars Surface Mission (Minimal)	Enhancing	2033	--	2027	2 years

14.1 Cryogenic Systems
14.1.1 Passive Thermal Control

14.1.1.7 Low Temperature Radiators

TECHNOLOGY

Technology Description: Spacecraft radiators that have high heat rejection at very low temperatures.

Technology Challenge: Challenges include testing at very low temperatures.

Technology State of the Art: Coatings that have high emissivity at temperatures below 50 K have been developed. These need to be integrated with low-mass radiator panels and heat pipes that can operate in low temperatures.

Parameter, Value:
Emissivity: 0.93 at 20 K
Mass per unit area: low
Structurally adequate for spaceflight

TRL 5

Technology Performance Goal: Increase emissivity with lower mass.

Parameter, Value:
Emissivity greater than 0.9 in the 50 K to 20 K range.

TRL 7

Technology Development Dependent Upon Basic Research or Other Technology Candidate: None

CAPABILITY

Needed Capability: High heat rejection at very low temperatures.

Capability Description: Provide high heat rejection capability for cooling below 50 K with higher heat loads.

Capability State of the Art: Typically, heat is lifted to a higher temperature using a cryocooler, and the heat is then rejected by a radiator at that higher temperature.

Parameter, Value:
Emissivity: significant drop at temperatures below 100 K.

Capability Performance Goal: Radiators with high infrared radiative surfaces combined with cooling loops that operate at temperatures below 50 K.

Parameter, Value:
Emissivity greater than 0.9 in the 50 K to 20 K range.

Technology Needed for the Following NASA Mission Class and Design Reference Mission	Enabling or Enhancing	Mission Class Date	Launch Date	Technology Need Date	Minimum Time to Mature Technology
Exploring Other Worlds: DRM 8 Crewed to Mars Moons	Enhancing	2027	2027	2021	2 years
Planetary Exploration: DRM 8a Crewed Mars Orbital	Enhancing	2033	--	2027	2 years
Planetary Exploration: DRM 9 Crewed Mars Surface Mission (DRA 5.0)	Enhancing	2033	--	2027	2 years
Planetary Exploration: DRM 9a Crewed Mars Surface Mission (Minimal)	Enhancing	2033	--	2027	2 years

14.1 Cryogenic Systems
14.1.1 Passive Thermal Control

14.1.1.8 Cryogenic Heat Pipes

TECHNOLOGY

Technology Description: Heat pipes and heat spreaders that are effective at temperatures below 50 K. Specialized applications require devices that operate below 4 K.

Technology Challenge: Challenges include obtaining adequate conductance.

Technology State of the Art: Small-scale demonstrations have occurred to date.

Parameter, Value:	TRL
1.75 W at 16 K.	4

Technology Performance Goal: Increase conductance for helium, hydrogen, and neon.

Parameter, Value:	TRL
High conductance (W/mK) at operating temperatures of 4 K for helium, 15 K for hydrogen, and 24 K neon.	6

Technology Development Dependent Upon Basic Research or Other Technology Candidate: Thermodynamic and fluid dynamic behavior (liquid-vapor) of cryogenic fluids in partial or microgravity.

CAPABILITY

Needed Capability: Cooling transport for specialized applications that operate at cryogenic temperatures.

Capability Description: Provide cooling transport systems for helium, hydrogen, and neon that are effective at temperatures below 50 K. Specialized applications require devices that operate below 4 K.

Capability State of the Art: Current heat pipes are limited to operation above the freezing point of the working fluid.

Parameter, Value:

Operating temperature: > 195 K for ammonia and > 84 K for propylene.

Capability Performance Goal: Need helium, hydrogen, and neon heat pipes capable of providing cooling transport at 4 K, 15 K, and 40 K, respectively.

Parameter, Value:

High conductance (W/mK) at operating temperatures of 4 K for helium, 15 K for hydrogen, and 24 K neon.

Technology Needed for the Following NASA Mission Class and Design Reference Mission	Enabling or Enhancing	Mission Class Date	Launch Date	Technology Need Date	Minimum Time to Mature Technology
Strategic Missions: Far Infrared Surveyor Mission	Enhancing	--	2035*	2035	4 years

*Launch date is estimated and not in Agency Mission Planning Model (AMPM)

14.1 Cryogenic Systems
14.1.2 Active Thermal Control

14.1.2.1 High Capacity 20 Kelvin Cryocoolers

TECHNOLOGY

Technology Description: Spaceflight cryocooler for cooling liquid hydrogen propellant tanks.

Technology Challenge: Challenges include developing a recuperator fabrication process and integrating the recuperator, compressor, and turbo-alternator.

Technology State of the Art: A spaceflight-qualified cryocooler demonstrated a cooling capacity 6 W at 35 K.

Parameter, Value:

Cooling capacity: 6.3 W at 49 K
Specific power: 53 W/W
Specific mass: 1.9 kg/W

TRL 4

Technology Performance Goal: 20 K cryocooler

Parameter, Value:

Cooling capacity: greater than 20 W at 20 K
Specific power: less than 80 W/W
Specific mass: less than 5 kg/W

TRL 6

Technology Development Dependent Upon Basic Research or Other Technology Candidate: None

CAPABILITY

Needed Capability: Cooling capacity to achieve zero boil-off of liquid hydrogen.

Capability Description: Provide high-capacity 20 K cryocoolers to achieve zero boil-off for liquid hydrogen storage, which enables the use of hydrogen propellant for long-duration space exploration.

Capability State of the Art: The current spaceflight technology does not provide the required cooling capacity.

Parameter, Value:

Cooling capacity is 1 W at 20 K with a specific power of 200 W/W.

Capability Performance Goal: 20 K cryocooler capable of providing efficient cooling to achieve zero boil-off of large liquid hydrogen propellant tanks.

Parameter, Value:

Cooling capacity: greater than 20 W at 20 K
Specific power: less than 80 W/W
Specific mass: less than 5 kg/W

Technology Needed for the Following NASA Mission Class and Design Reference Mission	Enabling or Enhancing	Mission Class Date	Launch Date	Technology Need Date	Minimum Time to Mature Technology
Exploring Other Worlds: DRM 6 Crewed to NEA	Enabling	2027	2027	2021	2 years
Exploring Other Worlds: DRM 7 Crewed to Lunar Surface	Enabling	2027	2027	2021	2 years
Exploring Other Worlds: DRM 8 Crewed to Mars Moons	Enabling	2027	2027	2021	2 years
Planetary Exploration: DRM 8a Crewed Mars Orbital	Enabling	2033	--	2027	2 years
Planetary Exploration: DRM 9 Crewed Mars Surface Mission (DRA 5.0)	Enabling	2033	--	2027	2 years
Planetary Exploration: DRM 9a Crewed Mars Surface Mission (Minimal)	Enabling	2033	--	2027	2 years

14.1 Cryogenic Systems
14.1.2 Active Thermal Control

14.1.2.2 High Capacity 90 Kelvin Cryocoolers

TECHNOLOGY

Technology Description: Spaceflight cryocooler for cooling liquid oxygen propellant tanks and for broad area cooling (BAC) shield for liquid hydrogen tanks.

Technology Challenge: Challenges include developing a recuperator fabrication process and integrating the recuperator, compressor, and turbo-alternator.

Technology State of the Art: A flight-representative reverse turbo Brayton cycle cryocooler has been demonstrated 15 W at 90 K.

Parameter, Value:	TRL
Cooling capacity: 15 W at 90 K Specific power: 22 W/W Specific mass: 3.3 kg/W	4

Technology Performance Goal: 90 K cryocooler.

Parameter, Value:	TRL
Cooling capacity: greater than 150 W at 90 K Specific power: less than 10.6 W/W Specific mass: less than 0.35 kg/W	6

Technology Development Dependent Upon Basic Research or Other Technology Candidate: None

CAPABILITY

Needed Capability: Cooling capacity at 90 K to reduce propellant boil-off.

Capability Description: Provide high-capacity 90 K cryocoolers to achieve zero boil-off for liquid oxygen or liquid methane propellant, reduce boil-off for liquid hydrogen, and reduce the required cooling capacity of the 20 K cooler needed to achieve zero boil-off for liquid hydrogen.

Capability State of the Art: Flight-qualified pulse tube cryocooler has been demonstrated in a space-like environment.

Parameter, Value:

Cooling capacity: 10W at 90 K

Specific power: 14 W/W

Specific mass: 0.4 kg/W

Capability Performance Goal: 90 K cryocooler capable of providing efficient cooling to achieve reduced boil-off of large liquid hydrogen propellant tanks.

Parameter, Value:

Cooling capacity: greater than 150 W at 90 K

Specific power: less than 10.6 W/W

Specific mass: less than 0.35 kg/W

Technology Needed for the Following NASA Mission Class and Design Reference Mission	Enabling or Enhancing	Mission Class Date	Launch Date	Technology Need Date	Minimum Time to Mature Technology
Exploring Other Worlds: DRM 6 Crewed to NEA	Enhancing	2027	2027	2021	2 years
Exploring Other Worlds: DRM 7 Crewed to Lunar Surface	Enhancing	2027	2027	2021	2 years
Exploring Other Worlds: DRM 8 Crewed to Mars Moons	Enhancing	2027	2027	2021	2 years
Planetary Exploration: DRM 8a Crewed Mars Orbital	Enhancing	2033	--	2027	2 years
Planetary Exploration: DRM 9 Crewed Mars Surface Mission (DRA 5.0)	Enhancing	2033	--	2027	2 years
Planetary Exploration: DRM 9a Crewed Mars Surface Mission (Minimal)	Enhancing	2033	--	2027	2 years

14.1 Cryogenic Systems
14.1.2 Active Thermal Control

14.1.2.3 High Capacity Cryocoolers for In-Situ Manufacture of Cryogenic Fluids

TECHNOLOGY

Technology Description: High-capacity, high-efficiency, low-mass cryocoolers for liquid oxygen and liquid methane production on the Martian surface, and liquid oxygen or methane production on the lunar surface.

Technology Challenge: Challenges include qualifying commercially-available cryocoolers for spaceflight and developing space-qualified controllers.

Technology State of the Art: Industrial cryocoolers can be modified for space applications.

Parameter, Value:

Cooling capacity at 77 K: 16 W

Specific power: 15 W/W

Specific mass: 0.2 kg/W

Percent of Carnot efficiency: 20%

TRL 3

Technology Performance Goal: Provide cryocoolers with ability to capture, purify, and compress Martian atmospheric gases for processing.

Parameter, Value:

Rate: 12.1 kg CO_2/hr to produce 2.2 kg O_2/hr.

TRL 6

Technology Development Dependent Upon Basic Research or Other Technology Candidate: None

CAPABILITY

Needed Capability: Provide cooling to freeze out Mars (CO_2) atmosphere stream.

Capability Description: Provide cryocoolers with the ability to capture, purify, and compress Martian atmospheric gases for processing.

Capability State of the Art: Flight-qualified cryocoolers with low capacity.

Parameter, Value:

Cooling capacity at 77 K: 16 W

Specific power: 15 W/W

Specific mass: 0.2 kg/W

Percent of Carnot efficiency: 20%

Capability Performance Goal: Need 2.2 kg O_2/hr for Mars crew ascent. Cryocooler needs to produce a certain amount of cooling capacity to freeze CO_2 out of the Mars atmosphere. Value given a very rough approximation based on calculations for much smaller scales. Efficiency is based on electrical power required to produce cooling capacity.

Parameter, Value:

Rate: 12.1 kg CO_2/hr (to produce 2.2 kg O_2/hr for Mars crew ascent. Purity depends on downstream processing method; Sabatier, reverse water gas shift, and solid oxide electrolysis might all have different sensitivity to other gases in the feed stream.

Pressure: 15 – 40 psia

Technology Needed for the Following NASA Mission Class and Design Reference Mission	Enabling or Enhancing	Mission Class Date	Launch Date	Technology Need Date	Minimum Time to Mature Technology
Planetary Exploration: DRM 9 Crewed Mars Surface Mission (DRA 5.0)	Enabling	2033	--	2027	4 years

14.1 Cryogenic Systems
14.1.2 Active Thermal Control

14.1.2.4 Low-T, Low-Q Cryocoolers

TECHNOLOGY

Technology Description: Multi-stage, sub-Kelvin cryocoolers with high efficiency, and 4 K cryocoolers as heat sinks.

Technology Challenge: Challenges include vefirying thermal systems in non-ideal test chambers.

Technology State of the Art: The Continuous Adiabatic Demagnetization Refrigerator (CADR) is under development to enable cooling at 30 mK.

Parameter, Value:

The James Webb Space Telescope cooler can be upgraded to provide cooling at 4 K by substituting Helium-3 for Helium-4. CADR has demonstrated a cooling capacity of 6 µW at 50 mK and 1.5 µW at 35 mK, with a 5 K heat sink.

TRL 4

Technology Performance Goal: Multi-stage, sub-Kelvin cryocoolers cryogens.

Parameter, Value:

72mW at 18 K and 180 mW at 4 K with less than 200W of input power.

5 µW at 50 mK or 1µW at 30 K with a 4 K - 6 K heat sink. 5 K – 1.2 K, 1.2 K – 0.3 K, 0.3 K to 0.05 K, and 1.2 K. Input power 30-80W.

TRL 6

Technology Development Dependent Upon Basic Research or Other Technology Candidate: Thermodynamic and fluid dynamic behavior (liquid- vapor) of cryogenic fluids in partial or microgravity.

CAPABILITY

Needed Capability: Cryocooling for far-infrared (IR) and X-ray sensors.

Capability Description: Multi-stage, sub-Kelvin cryocoolers are required to achieve science goals for future far-IR telescopes or interferometers. Space-qualified 4 K cryocoolers will replace expendable cryogens. Adiabatic demagnetization refrigerators (ADRs) are just one cryocooler technology needing improvement.

Capability State of the Art: Numerous flight cryocoolers are listed in the Cryocoolers Conference Proceding series, edited by R.G. Ross. Also, consumable liquid hydrogen is used, which is a huge consumer of volume and mass, drives mission cost, and limits mission lifetime.

Parameter, Value:

Plank hydrogen sorption cooler

Cooling capacity: 230 mW at 18 K

Input power: 550 W

Compressor volume: 0.16 m^2

Compressor mass: 40 kg

JWST cooling capacity: 60 mW at 6 K, 75 mW at 18 K

Capability Performance Goal: Far-IR telescopes need cryocoolers that provide cooling at 18 K, 4 K, and at sub-1 K temperatures.

Parameter, Value:

Modified JWST cooling capacity: 72 mW at18 K and 180 mW at 4 K, with less than 200 W of input power.

Technology Needed for the Following NASA Mission Class and Design Reference Mission	Enabling or Enhancing	Mission Class Date	Launch Date	Technology Need Date	Minimum Time to Mature Technology
Strategic Missions: X-ray Surveyor Mission	Enhancing	--	2035*	2030	4 years
Strategic Missions: Far Infrared Surveyor Mission	Enabling	--	2035*	2035	4 years

*Launch date is estimated and not in Agency Mission Planning Model (AMPM)

14.1 Cryogenic Systems
14.1.2 Active Thermal Control

14.1.2.5 Distributed Cooling Loops

TECHNOLOGY

Technology Description: Cooling systems that reduce boil-off from cryogenic propellant tanks. May include cryogenic circulators, heat traps, and heat exchangers.

Technology Challenge: Challenges include developing an optimized distributed cooling loop system that takes into account mass, propellant loss, power, and other factors.

Technology State of the Art: Ground testing has demonstrated zero boil-off (ZBO) with a liquid oxygen tank, reduced boil-off (RBO) with a liquid hydrogen tank.

Parameter, Value:

LH_2 RBO reduction: 66%

LOX ZBO reduction: 100%

Mass per unit area: 1.2 kg/m^2

Pressure drop: 0.4 PSID

TRL

5

Technology Performance Goal: Minimize propellant loss and boil-off.

Parameter, Value:

RBO reduction: 66%

ZBO reduction: 100%

Mass per unit area: less than 1.2 kg/m^2

Pressure drop: low

TRL

6

Technology Development Dependent Upon Basic Research or Other Technology Candidate: None

CAPABILITY

Needed Capability: Systems capable of circulating cryogenic vapor.

Capability Description: Distribute cooling boil-off vapor or circulated cryogenic gas over large surfaces within the insulation and to discrete locations, such as piping or structural supports that penetrate the insulation.

Capability State of the Art: A small-scale cryogenic circulating system has flown on the Hubble telescope.

Parameter, Value:

There are a number of related parameters that lead to optimizing the system. These include mass, power, effectiveness, and the ability to meet minimum structural requirements.

Capability Performance Goal: RBO applications are to reduce propellant loss and ZBO applications are to eliminate boil-off.

Parameter, Value:

RBO reduction: 66%

ZBO reduction: 100%

Mass per unit area: less than 1.2 kg/m^2

Pressure drop: low

Technology Needed for the Following NASA Mission Class and Design Reference Mission	Enabling or Enhancing	Mission Class Date	Launch Date	Technology Need Date	Minimum Time to Mature Technology
Exploring Other Worlds: DRM 6 Crewed to NEA	Enhancing	2027	2027	2021	2 years
Exploring Other Worlds: DRM 7 Crewed to Lunar Surface	Enhancing	2027	2027	2021	2 years
Exploring Other Worlds: DRM 8 Crewed to Mars Moons	Enhancing	2027	2027	2021	2 years
Planetary Exploration: DRM 8a Crewed Mars Orbital	Enhancing	2033	--	2027	2 years
Planetary Exploration: DRM 9 Crewed Mars Surface Mission (DRA 5.0)	Enhancing	2033	--	2027	2 years
Planetary Exploration: DRM 9a Crewed Mars Surface Mission (Minimal)	Enhancing	2033	--	2027	2 years

14.1 Cryogenic Systems
14.1.2 Active Thermal Control

14.1.2.6 Pumps, Circulators, and Fans

TECHNOLOGY

Technology Description: Devices for transporting cryogenic liquids and gases are needed for many applications, such as propellant tank mixing, cryogenic fluid transfer, and broad area cooling (BAC) loops.

Technology Challenge: Challenges include developing pumping devices for transporting cryogen liquids and vapors that are low mass and highly efficient.

Technology State of the Art: Recent progress has been made in developing pumps with superconducting motors.

Parameter, Value:

Applicable capacity
Efficiency: high
Mass: low
Heat introduction: low
Reliability: high

TRL 4

Technology Performance Goal: Provide pumping and mixing technologies that are highly reliable and generate very low heat loads.

Parameter, Value:

No heat conducted to cryogenic fluid.

TRL 6

Technology Development Dependent Upon Basic Research or Other Technology Candidate: None

CAPABILITY

Needed Capability: Efficient cryogenic pumping devices .

Capability Description: Provide pumping and mixing technologies that are highly reliable and generate very low heat loads.

Capability State of the Art: Commercial/industrial cryogenic pumps with too much capacity, low efficiency, high heat loss, and high mass.

Parameter, Value:

Applicable capacity
Efficiency: high
Mass: low
Heat introduction: low
Reliability: high

Capability Performance Goal: Efficient pumping of cryogenic vapors and liquids, with minimal heat generated.

Parameter, Value:

Efficiency: high
Heat added: very low
Reliability: high

Technology Needed for the Following NASA Mission Class and Design Reference Mission	Enabling or Enhancing	Mission Class Date	Launch Date	Technology Need Date	Minimum Time to Mature Technology
Exploring Other Worlds: DRM 6 Crewed to NEA	Enhancing	2027	2027	2021	4 years
Exploring Other Worlds: DRM 7 Crewed to Lunar Surface	Enhancing	2027	2027	2021	4 years
Exploring Other Worlds: DRM 8 Crewed to Mars Moons	Enhancing	2027	2027	2021	4 years
Planetary Exploration: DRM 8a Crewed Mars Orbital	Enhancing	2033	--	2027	4 years
Planetary Exploration: DRM 9 Crewed Mars Surface Mission (DRA 5.0)	Enhancing	2033	--	2027	4 years
Planetary Exploration: DRM 9a Crewed Mars Surface Mission (Minimal)	Enhancing	2033	--	2027	4 years

14.1 Cryogenic Systems
14.1.2 Active Thermal Control

14.1.2.7 Integrated Radiator/Cryocooler for Liquefaction

TECHNOLOGY

Technology Description: High-capacity, high-efficiency, low-mass heat rejection system and controls for oxygen production on Martian surface.

Technology Challenge: Challenges include developing and flight qualifying an integrated system in a relevant environment.

Technology State of the Art: Integration of a heat pipe radiator with a reverse turbo Brayton cryocooler was recently demonstrated in a simulated space environment.

Parameter, Value:

Heat rejection at 300 K: 400 W

Minimizing mass was not addressed

Radiator efficiency: 0.98

TRL 5

Technology Performance Goal: Provide integrated radiator/cryocooler system for continuous liquefaction product stream at in-situ processing plant.

Parameter, Value:

Rate: liquefy 0.16 kg oxygen/hr for Lunar environment and 2.2 for Martian.

TRL 6

Technology Development Dependent Upon Basic Research or Other Technology Candidate: None

CAPABILITY

Needed Capability: Liquefaction and storage of oxygen.

Capability Description: Provide system for continuous liquefaction product stream at in-situ processing plant.

Capability State of the Art: A flight-qualified cryocooler heat rejection system has operated on the Hubble telescope and included capillary-pumped loop heat pipes.

Parameter, Value:

Heat rejected at 300 K: 400 W

Efficiency: High

Mass: Low

Capability Performance Goal: From Constellation lunar surface architecture needed 1000 kg oxygen per year for life support needs. Per DRM 7 - 1 crew mission per year sets storage duration. From DRM 9, ISRU plant pre-deployed one opportunity before crew and all oxygen is made and in tanks before crew leaves.

Parameter, Value:

Lunar:

Rate: liquefy 0.16 kg oxygen/hr

Duration: 1 year

Mars:

Rate: liquefy 2.2 kg oxygen/hr

Duration: greater than 3.5 years from first drop of O_2 produced

Technology Needed for the Following NASA Mission Class and Design Reference Mission	Enabling or Enhancing	Mission Class Date	Launch Date	Technology Need Date	Minimum Time to Mature Technology
Exploring Other Worlds: DRM 7 Crewed to Lunar Surface	Enabling	2027	2027	2021	6 years
Planetary Exploration: DRM 9 Crewed Mars Surface Mission (DRA 5.0)	Enabling	2033	--	2027	6 years

14.1 Cryogenic Systems
14.1.2 Active Thermal Control

14.1.2.8 Subcooling Cryogenic Propellants

TECHNOLOGY

Technology Description: Subcooling cryogenic propellants, such as liquid hydrogen (LH_2) or liquid oxygen (LO_2) prior to launch prevents boil-off for extended durations without adding launched mass. Subcooling can provide increased vent-free storage without added mass, enabling higher specific impulse rockets and fewer and less massive launches.

Technology Challenge: Subcooling prior to launch with a small physical footprint and subcooling prior to launch with a small amount of resources, such as electrical power and cryogens.

Technology State of the Art: LO_2 subcooling has been demonstrated using a reduced pressure cryogen bath (RPCB) with liquid nitrogen (LN_2) for a launch vehicle that uses large shell and tube heat exchanger and compressors.

LH_2 subcooling has been demonstrated over short durations using RPCB with LH_2 for a test article.

Parameter, Value:

LO_2 subcooling below 90K for large tanks (> 1000 L)

LH_2 subcooling using RPCB with LH_2 to 16 deg K for small scale test article.

TRL 3

Technology Performance Goal: Provide a compact reliable long life system for subcooling LH_2 on the launch pad:

< 1000 L propellant tanks: ground support equipment (GSE) cryocoolers;

1000 L -10,000 L propellant tanks: cryocoolers or thermodynamic cryogen subcooler (TCS) using uses J-T valve, two-phase heat exchanger, and compressors or RPCBs in a LH_2 bath;

> 10,000 L propellant tanks: TCS or RPCB.

Parameter, Value:

LH_2 subcooling to 16 deg K for long duration (> 24 hrs) for tanks > 10,000 L using TCS or RPCB

TRL 6

Technology Development Dependent Upon Basic Research or Other Technology Candidate: For reduced-pressure LH_2 bath and TCS subcooling, reliable, long-life H_2 compressors are necessary. These H_2 compressors are under development.

CAPABILITY

Needed Capability: Reliable, compact, and efficient subcooling systems for launch pad subcooling.

Capability Description: Subcooling cryogenic propellants prior to launch to prevent boil-off for extended durations without adding launched mass. For interplanetary missions, subcooling can provide years of vent-free hydrogen storage without additional launched mass, enabling the use of high-specific-impulse LH_2+LO_2 engines. For launch vehicle upper stages, subcooling can provide weeks to months of in-space, vent-free hydrogen storage (depending on parking orbits) without additional launched mass, allowing fewer launches and less massive launch vehicles.

Capability State of the Art: Commercially-available cryocoolers can be used for subcooling tanks that are < 1000 L. For bigger tanks, multiple coolers might be necessary or more cryocooler development is necessary. For tanks that are >10,000 L, it would be more efficient and compact to use TCS or RPCB. However, these require reliable compressors for extended use that do not exist.

Parameter, Value:

Commercially-available cryocoolers - Lifts 70 W at 20 K consuming 11.2 kW.

H_2 compressors for RPCB or TCS: < 1000 hours life.

Capability Performance Goal: Provide a compact, reliable, long-life system for subcooling LH_2 on the launch pad:

More compact and efficient GSE cryocoolers would be desirable.

More reliable H_2 compressors are necessary for RPCB and TCS.

Integrating subcooling hardware into the launch infrastructure.

Parameter, Value:

GSE or Brayton Cryocoolers: Lifts 100-600 W at 16 K consuming < 15-80 kW.

Compressors for RPCB or TCS: > 100,000 hours life

Technology Needed for the Following NASA Mission Class and Design Reference Mission	Enabling or Enhancing	Mission Class Date	Launch Date	Technology Need Date	Minimum Time to Mature Technology
Explorer Class: Explorer Missions	Enabling	--	2023	2020	3 years
New Frontiers: New Frontiers Program 4 (NF4/~2017 AO Release)	Enabling	--	2024	2016	2 years
Planetary Flagship: Europa	Enabling	--	2022*	2019	3 years

*Launch date is estimated and not in Agency Mission Planning Model (AMPM)

14.2 Thermal Control Systems
14.2.1 Heat Acquisition

14.2.1.1 Freeze Tolerant Heat Pipes

TECHNOLOGY

Technology Description: Heat transport devices that can freeze and thaw without damage or degraded performance.

Technology Challenge: Flight demonstration is required for constant conductance heat pipes due to uncertainties with basic thermophysics. However, if we use a passive variable conductance heat pipe, (VCHP) with an additional condenser length filled with non compressible gas (NCG), then the 0-G and 1-G performance will be similar. (i.e. flight demonstration is not required.) The real challenge is to find a convenient method to get the right amount of NCG charge.

Technology State of the Art: Freeze-tolerant designs for heat pipes in a gravity environment exist. Special designs are necessary for the 0-G environment. This may be accomplished by including a non-condensable gas and a volumetric expansion space.

Parameter, Value:

Absolute temperatures.

TRL

6

Technology Performance Goal: Provide two-phase heat transport devices that can undergo multiple freeze/thaw cycles without damage or degraded performance.

Parameter, Value:

Absolute temperatures and freeze/thaw cycle life.

TRL

9

Technology Development Dependent Upon Basic Research or Other Technology Candidate: Operating fluid, minimum thermal sink temperatures, selection of NCG.

CAPABILITY

Needed Capability: Heat transport technology capable of surviving repeated freeze/thaw cycles.

Capability Description: Provide heat pipes capable of sustained operation after being exposed to temperatures below the freeze point of their operating fluid.

Capability State of the Art: Conventional heat pipes with ammonia (the most common operating fluid) as an operating fluid can operate from about 210 K to 330 K. However, they may freeze when exposed to excessively cold temperatures, and survival is uncertain. The pipes may burst upon thawing or the internal capillary structure may be damaged. The same is true for heat pipes with different operating fluids, but the temperature limits are different.

Parameter, Value:

Survival temperature for heat pipes, which is specific to the operating fluid (e.g., freeze point). For ammonia, this is approximately 196 K.

Capability Performance Goal: Significantly lower survival temperature to cryogenic temperatures below 150 K for ammonia, and comparable improvements for other operating fluids.

Parameter, Value:

Two-phase heat transport device survival temperature: < 150 K.

Technology Needed for the Following NASA Mission Class and Design Reference Mission	Enabling or Enhancing	Mission Class Date	Launch Date	Technology Need Date	Minimum Time to Mature Technology
Exploring Other Worlds: DRM 6 Crewed to NEA	Enabling	2027	2027	2021	3 years
Discovery: Discovery 13	Enhancing	--	2020	2017	3 years
Exploring Other Worlds: DRM 8 Crewed to Mars Moons	Enabling	2027	2027	2021	3 years

14.2 Thermal Control Systems
14.2.1 Heat Acquisition

14.2.1.2 High Flux Heat Acquisition with Constant Temperature

TECHNOLOGY

Technology Description: Acquisition and removal of high heat fluxes (> 100W/cm²) over relatively small areas, like those for processor chips and laser diodes, with tight temperature control.

Technology Challenge: Flight demonstration is required over an extended time period (days to months) due to the effect of zero gravity in the space environment.

Technology State of the Art: Spray cooling with a single phase fluid is capable of heat removal at levels approaching 200 W/cm², but erosion and collection of spray effluent in zero gravity may be issues. Advanced heat pipes have demonstrated flux removal to only ~80 W/cm². Electro-hydrodynamic pumping of a coolant might achieve >100 W/cm² without the erosion or effluent collection issues.

Parameter, Value: Watts/cm²

TRL: 5

Technology Performance Goal: Provide heat acquisition and removal devices that can remove high flux levels of waste heat (not necessarily large quantities of heat, but possibly so) while maintaining tight temperature control. Electro-hydrodynamic pumping of a coolant might achieve >100 W/cm² without the erosion or effluent collection issues.

Parameter, Value: Watts/cm²

TRL: 9

Technology Development Dependent Upon Basic Research or Other Technology Candidate: Operating fluid, erosion of surfaces, and collection of condensate in micro-gravity.

CAPABILITY

Needed Capability: Heat acquisition devices capable of removing waste heat at high flux levels (> 100 W/cm²) while still maintaining temperatures within limits.

Capability Description: Provide devices capable of removing high flux waste heat while maintaining tight temperature control for integrated cicruit chips, laser heads, and other electronic components.

Capability State of the Art: Current technology for high flux heat removal includes high flow rates of a single phase fluid, vapor/liquid spray, or advanced heat pipes. These devices have limitations in maximum flux levels, degree of temperature control, and/or erosion of the surface. Also, demonstration of operation in zero gravity is needed.

Parameter, Value: Degree of temperature control (to +/- 1° C degrees) at high heat flux levels (> 100 W/cm²)

Capability Performance Goal: Provide stable, long-term high heat flux removal in a zero gravity environment.

Parameter, Value: Provide high heat flux removal (>100 W/cm²) with temperature control to +/- 1.0° C.

Technology Needed for the Following NASA Mission Class and Design Reference Mission	Enabling or Enhancing	Mission Class Date	Launch Date	Technology Need Date	Minimum Time to Mature Technology
Earth Systematic Missions: Active Sensing of CO_2 Emissions over Nights, Days, and Seasons (ASCENDS)	Enhancing	--	2023	2016	2 years

14.2 Thermal Control Systems
14.2.1 Heat Acquisition

14.2.1.3 Damage-Tolerant or Self-Healing Electric Heaters

TECHNOLOGY

Technology Description: Advanced electro-resistive materials, including nanotechnology, and component design.

Technology Challenge: Challenges include developing new materials.

Technology State of the Art		Technology Performance Goal	
Technology State of the Art: Electrically-resistive material sandwiched between two layers of insulating flexible film, such as Kapton.		**Technology Performance Goal:** Significantly reduced loss of heater functionality due to physical damage, delamination, or over-power.	
Parameter, Value:	**TRL**	**Parameter, Value:**	**TRL**
Mean time between failure	1	Mean time between failure	9

Technology Development Dependent Upon Basic Research or Other Technology Candidate: New materials, likely involving nanotechnology.

CAPABILITY

Needed Capability: Improved heater resistance to physical damage caused by various sources.

Capability Description: Enhance survival of electric heaters from over-temperature events or physical damage.

Capability State of the Art	Capability Performance Goal
Capability State of the Art: Protection for electric heaters is currently provided by over-temperature thermostats, multiple traces, and similar design features. This is a very mature technology.	**Capability Performance Goal:** Significantly improved tolerance to damage from space environment or over-temperature.
Parameter, Value:	**Parameter, Value:**
99% reliability over intended lifetime.	> 99% reliability.

Technology Needed for the Following NASA Mission Class and Design Reference Mission	Enabling or Enhancing	Mission Class Date	Launch Date	Technology Need Date	Minimum Time to Mature Technology
Strategic Missions: Push	Enhancing	--	--	--	3 years
Discovery: Push	Enhancing	--	--	--	3 years

14.2 Thermal Control Systems
14.2.1 Heat Acquisition

14.2.1.4 Insulation

TECHNOLOGY

Technology Description: Lightweight thermal insulators with low conductivity and effective emissivity (ε*).

Technology Challenge: Increasing insulative properties further requires development of new materials and insulation configurations.

Technology State of the Art: Areal blankets made by stacking multiple layers of low-emittance films, typically Mylar.

Parameter, Value:
Effective emissivity (ε*): 0.015 to 0.030

TRL
9

Technology Performance Goal: Improved effective emissivity of areal insulations.

Parameter, Value:
Effective emissivity (ε*): 0.005

TRL
9

Technology Development Dependent Upon Basic Research or Other Technology Candidate: None

CAPABILITY

Needed Capability: Minimize system heat transfer.

Capability Description: Minimize heat transfer between structural elements and from the spacecraft system to convective and radiative thermal environments.

Capability State of the Art: Large surface area thermal barrier between spacecraft components and structures and the external environment or local extreme temperature components.

Parameter, Value:
ε* value: 0.015 to 0.030

Capability Performance Goal: Significantly increased thermal performance.

Parameter, Value:
Lower ε* of flight blankets nearer to the theoretical value of 0.005.

Technology Needed for the Following NASA Mission Class and Design Reference Mission	Enabling or Enhancing	Mission Class Date	Launch Date	Technology Need Date	Minimum Time to Mature Technology
Strategic Missions: Push	Enhancing	--	--	--	3 years

14.2 Thermal Control Systems
14.2.2 Heat Transport

14.2.2.1 Heat Transport Fluid

TECHNOLOGY

Technology Description: Heat transport fluids that provide optimal thermo-physical properties for pumped loop acquisition, transport, and rejection while providing low toxicity to crew.

Technology Challenge: Current known fluids show optimal performance in only some of the performance parameters. The challenge is to identify the appropriate combination of performance and develop a fluid that can match that performance. This is a multi-dimensional trade space and there are many fluids to evaluate.

Technology State of the Art: Ground-based systems are not relevant to this technology. Uncrewed missions do not require non-toxic inner loop fluids. Current human spaceflight vehicles use dual-loop architecture with different fluids optimized for performance and crew toxicity.

Parameter, Value:

Fluid performance is assessed at the system level.

TRL

3

Technology Performance Goal: Operations using transport fluids at extremely low temperatures with thermal performance and toxicity of water.

Parameter, Value:

Specific heat; viscosity; freeze temperature; thermal conductivity; toxicity; and flammability.

TRL

6

Technology Development Dependent Upon Basic Research or Other Technology Candidate: Freeze/thaw phase change behavior in partial or microgravity.

CAPABILITY

Needed Capability: Optimized heat transport for space system thermal control.

Capability Description: Provide a heat transport fluid that can support a single-loop thermal transport architecture when combined with radiator turndown capability.

Capability State of the Art: Current human spaceflight vehicles use a dual-loop architecture that allows for use of high-performance but toxic fluids on the exterior loop and benign fluids within the crew enclosure. These systems require dual pumps and an interface heat exchanger. Designer heat transport fluids are required to optimize thermal transport loop performance for single-loop architectures when combined with turn-down radiator capability. Optimal thermal characteristics allow for high range of thermal environments and heat loads while providing low toxicity and flammability for crew safety.

Parameter, Value:

Dual loop system with two optimized thermal transport fluids.

Capability Performance Goal: Development of “designer” heat transport fluids that can support a single-loop thermal transport architecture, possibly when combined with radiator turndown capability. Elimination of interface heat exchanger and ability to service pump from crew cabin. Ability to operate TCS across mission requirements for thermal environment and heat loads. Ability to operate thermal control system across range of mission requirements for thermal environment and heat loads with increased reliability and reduced mass.

Parameter, Value:

Increase optimization of specific heat

Decrease viscosity

Decrease freeze temperature

Increase thermal conductivity

Decrease toxicity

Decrease flammability

Technology Needed for the Following NASA Mission Class and Design Reference Mission	Enabling or Enhancing	Mission Class Date	Launch Date	Technology Need Date	Minimum Time to Mature Technology
Into the Solar System: DRM 5 Asteroid Redirect – Crewed in DRO	Enabling	2022	2022	2015 - 2021	4 years

14.2 Thermal Control Systems
14.2.2 Heat Transport

14.2.2.2 Advanced Pumps

TECHNOLOGY

Technology Description: Long-life pumps for circulating heat transfer fluids.

Technology Challenge: Challenges include developing extra long life bearing and anti-corrosion technology.

Technology State of the Art: Mechanical pumps have been used in many ground and aeronautical applications for over a hundred years. The technology is very mature, with some special designs achieving very long lifetimes.

Parameter, Value: Very long pump lifetimes.

TRL: 9

Technology Performance Goal: Multiples of SOA (e.g., a goal of > 50,000 hours lifetime till failure) for flight applications.

Parameter, Value: Mean time between failure; hours.

TRL: 9

Technology Development Dependent Upon Basic Research or Other Technology Candidate: Materials, bearing design, contaminates, and environment (e.g., radiation, temperature).

CAPABILITY

Needed Capability: Long-life convective heat transfer.

Capability Description: Provide devices that can circulate heat transfer fluids with exceptionally long lifetimes.

Capability State of the Art: Reliable mechanical pumping of trichlorofluoromethane (CFC-11) refrigerant through a closed loop to provide cooling to instruments. Low-TRL (3) electrohydrodynamic pumping has been demonstrated.

Parameter, Value: Pump life: > 50,000 hours

Capability Performance Goal: Multiples of SOA (e.g., a goal of >50,000 hours) for flight applications in a relevant environment. Europa mission is a likely candidate. Additional requirements include radiation tolerance, acceptable pressure drop, and minimal power.

Parameter, Value: Mean time between failure; hours.

Technology Needed for the Following NASA Mission Class and Design Reference Mission	Enabling or Enhancing	Mission Class Date	Launch Date	Technology Need Date	Minimum Time to Mature Technology
Discovery: Push	Enhancing	--	--	--	3 years

14.2 Thermal Control Systems
14.2.2 Heat Transport

14.2.2.3 Heat Straps

TECHNOLOGY

Technology Description: Mechanical devices with extremely high thermal conductivity.

Technology Challenge: Challenges include improving thermal conductivity per specific mass, while maintaining suitability for space environment (e.g., vibration).

Technology State of the Art: Flexible straps made of a highly conductive metal, such as aluminum or copper, possibly with encapsulated carbon cores.

Parameter, Value: Heat transferred per degree of temperature (W/K).

TRL 9

Technology Performance Goal: Improve heat transferred per degree of temperature.

Parameter, Value: Heat transferred per degree of temperature (W/K): 50% improvement

TRL 9

Technology Development Dependent Upon Basic Research or Other Technology Candidate: Materials and design of end fittings.

CAPABILITY

Needed Capability: Extremely high conductive heat transfer between spacecraft components and systems over relatively long distances.

Capability Description: Extremely high conductive heat transfer straps with enhanced heat transport via conduction over a distance.

Capability State of the Art: Heat straps allow for heat transport via conduction over a distance. Conventionally, they are made from copper or aluminum, but new designs are made with carbon-based composites.

Parameter, Value: Heat transferred per degree of temperature (W/K) - 0.82 W/K for room temperature applications.

Capability Performance Goal: 50% improvement over SOA. Some low-TRL Small Business Innovative Research (SBIR) work has been done on low-weight, high-conductivity thermal straps made from high-conductivity carbon, but additional work is needed.

Parameter, Value: Heat transferred per degree of temperature (W/K) – 1.23 W/K for room temperature applications.

Technology Needed for the Following NASA Mission Class and Design Reference Mission	Enabling or Enhancing	Mission Class Date	Launch Date	Technology Need Date	Minimum Time to Mature Technology
Exploring Other Worlds: DRM 6 Crewed to NEA	Enhancing	2027	2027	2021	3 years
Strategic Missions: Push	Enhancing	--	--	--	3 years

14.2 Thermal Control Systems
14.2.2 Heat Transport

14.2.2.4 Heat Switches

TECHNOLOGY

Technology Description: Remotely-actuated mechanical devices with high thermal conductivity on/off ratio.

Technology Challenge: Challenges include improving thermal conductance, mass, and lifetime.

Technology State of the Art: Materials with differential coefficient of thermal expansion as a function of temperature allow contraction and expansion to create physical contact between materials for on/off actuation. Two-phase heat pipe devices can be made to create on/off heat transport switches.

Parameter, Value:

Closed-to-open heat conduction ratio: 100:1

Mass: 100 g

Lifetime: 1000 cycles

TRL

9

Technology Performance Goal: Higher on/off thermal conductivity ratio, descreased mass/volume (one-fifth current), wider temperature rante and 2 times greater reliability.

Parameter, Value:

Closed-to-open heat conduction ratio: 10 time greater than SOA

Mass: 20 g

Lifetime: 2000 cycles

TRL

9

Technology Development Dependent Upon Basic Research or Other Technology Candidate: Materials and mechanical design.

CAPABILITY

Needed Capability: At least 10 times higher on/off thermal conductivity ratio, 5 times lower mass/volume, wider temperature range, and 2 times higher reliability/lifetime.

Capability Description: A heat switch will “close” to allow heat conduction and “open” to retard heat conduction in order to help ensure proper temperature regulation. Control may be passive or active. Close-to-open conductance ratios of 100:1 are typical for mechanical devices.

Capability State of the Art: A heat switch will “close” to allow heat conduction and “open” to retard heat conduction in order to help insure proper temperature regulation. Control may be passive or active. Close-to-open conductance ratios of 100:1 are typical.

Parameter, Value:

Closed-to-open heat conduction ratio: 100:1

Mass: 100g

Lifetime: 1000 cycles

Capability Performance Goal: 2 times improvement over SOA in on/off switching ratio

For gas-gap switches used in cryogenic instrument applications, the on/off performance goal is 10:000:1.

Parameter, Value:

On/off switching ratio: 1000:1

Mass: 20g

Lifetime: 2000 cycles

Technology Needed for the Following NASA Mission Class and Design Reference Mission	Enabling or Enhancing	Mission Class Date	Launch Date	Technology Need Date	Minimum Time to Mature Technology
Strategic Missions: Push	Enhancing	--	--	--	3 years

14.2 Thermal Control Systems
14.2.2 Heat Transport

14.2.2.5 Heat Pipe Capillary Based Loops

TECHNOLOGY

Technology Description: Closed, two-phase heat transfer loops serving multiple heat loads and rejecting to multiple thermal sinks, with tight temperature control and minimal temperature drops.

Technology Challenge: Flight hardware has been built and ground tested. A flight experiment is needed.

Technology State of the Art: Conventional heat pipe loops are limited to a single evaporator and single condenser.

Parameter, Value:	TRL
Number of evaporators: 1 Number of condensers: 1	6

Technology Performance Goal: Achieve heat pipe loops with multiple evaporators and multiple condensers.

Parameter, Value:	TRL
Number of evaporators: > 1 Number of condensers: > 1	9

Technology Development Dependent Upon Basic Research or Other Technology Candidate: Fluid boiling/evaporation and condensation phase change behavior in partial or microgravity.

CAPABILITY

Needed Capability: Advanced two-phase heat transfer.

Capability Description: Provide heat pipe capillary loops capable of acquiring heat from multiple sources, transporting and sharing this heat, and then rejecting heat from multiple condensers.

Capability State of the Art: Capillary forces developed in a wick located only in the evaporator drive heat pipe loops. These loops, called either loop heat pipes or capillary pumped loops, differ from traditional heat pipes in that there is a separate liquid-filled line from the condenser to the evaporator, and a separate vapor-filled line from the evaporator to the condenser. Heat pipe loops can transport heat over exceptionally large distances (to tens of meters) with negligible temperature drop (tenths of a degree C). They may transport small or large quantities of heat (ratios of 1:1000 are possible) and may control temperature to a few 1/10s of a degree C. They offer several orders of magnitude improvement over heat pipes.

Parameter, Value:

Number of evaporators: 1

Number of condensers: 1

Capability Performance Goal: Achieve heat pipe loops with multiple evaporators and multiple condensers to acquire heat from multiple sources within a system and reject the heat to one or more sinks. Design concepts suitable for cryogenic operating fluids are also desired.

Parameter, Value:

Number of evaporators: > 1

Number of condensers: > 1

Technology Needed for the Following NASA Mission Class and Design Reference Mission	Enabling or Enhancing	Mission Class Date	Launch Date	Technology Need Date	Minimum Time to Mature Technology
Exploring Other Worlds: DRM 6 Crewed to NEA	Enhancing	2027	2027	2021	3 years
Exploring Other Worlds: DRM 7 Crewed to Lunar Surface	Enhancing	2027	2027	2021	3 years
Planetary Exploration: DRM 8a Crewed Mars Orbital	Enhancing	2033	--	2027	3 years

14.2 Thermal Control Systems
14.2.2 Heat Transport

14.2.2.6 Heat Pump

TECHNOLOGY

Technology Description: Devices that use energy to transfer heat against a thermal gradient to reject heat to a higher temperature sink.

Technology Challenge: Challenges include increasing heat pump efficiency.

Technology State of the Art: Well-developed refrigeration industry on Earth (gravity dependent); The International Space Station (ISS) has a refrigerated centrifuge and low-efficiency thermoelectric refrigerators. Shuttle had limited-life refrigerators.

Parameter, Value: 40% of Carnot coefficient of performance (COP)

TRL: 4

Technology Performance Goal: 50% of Carnot coefficient of performance (COP), micro-gravity compatible.

Parameter, Value: 50% of Carnot COP.

TRL: 7

Technology Development Dependent Upon Basic Research or Other Technology Candidate: Fluid boiling/evaporation and condensation phase change behavior in partial or microgravity.

CAPABILITY

Needed Capability: Reject heat to relatively hotter thermal environments.

Capability Description: Reject heat near the lunar equator at solar noon.

Capability State of the Art: Heat pumps use energy to provide thermal lift to move heat against a temperature gradient or to increase heat transfer via radiators with a thermal gradient. Alternatives include shading devices or heat storage (temporary only).

Parameter, Value: Percent of COP. Carnot COP is the limit of heat transferred against the thermal gradient per unit of energy input.

Capability Performance Goal: Reject spacecraft systems heat near the lunar equator at solar noon.

Parameter, Value: Percent of COP with power reduction, mass reduction, and reliability increase.

Technology Needed for the Following NASA Mission Class and Design Reference Mission	Enabling or Enhancing	Mission Class Date	Launch Date	Technology Need Date	Minimum Time to Mature Technology
Exploring Other Worlds: DRM 7 Crewed to Lunar Surface	Enhancing	2027	2027	2021	6 years

14.2 Thermal Control Systems
14.2.2 Heat Transport

14.2.2.7 Thermal Electric Coolers

TECHNOLOGY

Technology Description: Solid-state device that uses the Peltier effect to pump heat against a thermal gradient.

Technology Challenge: Challenges include developing more efficient thermal electric coolers (TECs) that operate at lower temperatures.

Technology State of the Art: Current efficiencies of around 5% are typical, and this decreases at lower operating temperature. The lowest usable operating temperature is about 150 K with a three-stage TEC.

Parameter, Value:
Efficiency: 5%
Operating temperature: 150 K

TRL 4

Technology Performance Goal: Improvement in efficiency at comparable sink temperatures, and decreased operating temperatures.

Parameter, Value:
Efficiency: 15% improvement
Operating temperature: 80 K to 100 K

TRL 9

Technology Development Dependent Upon Basic Research or Other Technology Candidate: Materials, perhaps nanotechnology, assembly techniques, and fundamental physics.

CAPABILITY

Needed Capability: Solid-state heat pump.

Capability Description: Provide a spot cooling device for sensors, electronics, and thermal control applications, with 10% to 15% efficiency increase and reliable operation down to 80 – 100K from current devices.

Capability State of the Art: TECs are solid-state devices that use the Peltier effect wherein an electric voltage potential across a junction of two different materials can create a cooling or heating effect, depending on the direction of the current. Address performance and mechanical reliability to 80 to 90 K.

Parameter, Value:
Efficiency: 5%
Operating temperature: 150 K

Capability Performance Goal: Both improved efficiencies at comparable absolute temperatures and differentials and operation at lower temperatures are desired. Consideration for specific sink temperature for particular applications must be made.

Parameter, Value:
Efficiency: 10% to 15% increase
Operating temperatures: 80 K to 90 K

Technology Needed for the Following NASA Mission Class and Design Reference Mission	Enabling or Enhancing	Mission Class Date	Launch Date	Technology Need Date	Minimum Time to Mature Technology
Strategic Missions: Push	Enhancing	--	--	--	3 years

14.2 Thermal Control Systems
14.2.2 Heat Transport

14.2.2.8 In-Situ Thermal Fluids Chemical Analysis

TECHNOLOGY

Technology Description: In-situ thermal fluids chemical analysis to monitor thermal transport fluid health status.

Technology Challenge: Challenges include developing an autonomous, low-volume, and low-power sampling and analysis capability.

Technology State of the Art: Current health monitoring of the International Space Station (ISS) thermal transport fluids and system requires return of fluid samples to Earth for analysis with system treatment capability provided via on-orbit kit or launched to the ISS.

Parameter, Value: Analysis of fluid health is performed on the ground.

TRL: 3

Technology Performance Goal: Technology should allow for real-time measurement and analysis of transport fluid chemical properties and command required treatment dosage.

Parameter, Value: Accuracy of measurement and analysis, maintenance of thermal transport fluid properties within specified parameters.

TRL: 6

Technology Development Dependent Upon Basic Research or Other Technology Candidate: Sensor packaging and miniaturization and a tailored sensor capability.

CAPABILITY

Needed Capability: Real-time mission knowledge of thermal fluids health status.

Capability Description: Provide onboard chemical analysis of thermal control system fluids.

Capability State of the Art: Thermal control fluids health monitoring provides real-time system performance status and allows crew and ground personnel to take action to assure continued operation. Understanding thermal transport fluid health through in-situ chemical analysis provides insight into when fluid replacement or treatment is needed.

Parameter, Value: Accuracy, timeliness, and cost of acquisition of fluid health data.

Capability Performance Goal: Provide real-time, low-volume, high-reliability monitoring of thermal control system fluid health.

Parameter, Value: Accuracy, timeliness, and cost of acquisition of fluid health data.

Technology Needed for the Following NASA Mission Class and Design Reference Mission	Enabling or Enhancing	Mission Class Date	Launch Date	Technology Need Date	Minimum Time to Mature Technology
Into the Solar System: DRM 5 Asteroid Redirect – Crewed in DRO	Enhancing	2022	2022	2015 - 2021	5 years

14.2 Thermal Control Systems
14.2.2 Heat Transport

14.2.2.9 High-Thermal-Conductivity Thermal Interface Materials

TECHNOLOGY

Technology Description: Advanced materials that can provide very high thermal interface conductance but also be workable. These must be vacuum compatible and must not degrade performance through repeated cycling.

Technology Challenge: Challenges include developing new materials that are vacuum compatible, rugged, and have high thermal conductivity.

Technology State of the Art: A wide variety of thermal interface materials have been developed, including greases, thin metal foils, room temperature vulcanizing (RTV) polymers. In addition to high thermal conductance it is important for such interface material to be removable/reusable, have repeatable and predictive properties, negligable outgassing, be highly compliant, and perform well at nominal contact pressures and temperatures.

Parameter, Value: Thermal conductance: ~0.5 to 1.0 W/in^2-°C at room temperature

TRL: 6

Technology Performance Goal: Increase thermal conductance at room temperature and nominal pressures with comparable reusability, negligable outgasing, and compliance. Nominal variation from cryo to hot temperatures is also important.

Parameter, Value: Thermal conductance: 2.0 W/cm-°C at room temperature

TRL: 9

Technology Development Dependent Upon Basic Research or Other Technology Candidate: Materials, installation techniques, area of application, and clamping force.

CAPABILITY

Needed Capability: High-conductivity and reusable heat transfer interface material.

Capability Description: Provide advanced materials that can improve thermal interface conductance but also be capable of multiple make/break cycles.

Capability State of the Art: SOA is highly dependent upon the specific application drivers, with typical conductance values of approximately 0.5 to 1.0 W/in^2-°C @ room temperature, as measured in the center of interface plate. New generations of electronics used on numerous missions have much higher power densities than those used in the past.

Parameter, Value: Interface conductance (W/in^2-°C): 0.5 to 1.0 W/in^2-°C @ room temperature

Capability Performance Goal: Improve thermal conductivity across the interface by in the near and far terms. (NOTE: thermal conductivity is always a strong function of applied pressure). Reduce interface temperature gradients and facilitate efficient heat removal, high-conductivity using vacuum-compatible interface materials that minimize losses across make/break interfaces. The new interface material should also provide improved cyclic durability and ease of re-workability.

Parameter, Value: Interface conductance (W/in^2-°C): Factor of 2 increase in the near term and facter of 10 longer term.

Technology Needed for the Following NASA Mission Class and Design Reference Mission	Enabling or Enhancing	Mission Class Date	Launch Date	Technology Need Date	Minimum Time to Mature Technology
Earth Systematic Missions: Active Sensing of CO_2 Emissions over Nights, Days, and Seasons (ASCENDS)	Enhancing	--	2023	2016	2 years
Earth Systematic Missions: Climate Absolute Radiance and Refractivity Observatory (CLARREO)	Enhancing	--	2021*	2016	2 years
Earth Systematic Missions: Aerosol-Cloud-Ecosystems (ACE)	Enhancing	--	2024*	2020	2 years

*Launch date is estimated and not in Agency Mission Planning Model (AMPM)

14.2 Thermal Control Systems
14.2.2 Heat Transport

14.2.2.10 Micro- and Nano-Scale Heat Transfer Surfaces

TECHNOLOGY

Technology Description: Develop advanced heat transfer surfaces and flow channels that incorporate combined micro-scale (μm-mm) and nano-scale (nm) features to enhance two-phase flow heat transfer with higher heat fluxes and enhanced flow stability.

Technology Challenge: Challenges include achieving ultra-high heat flux capability (>1000 W/cm^2) with tight temperature control for phase change heat exchangers, two-phase pumped loops, and heat pipe capillary loops, as well as heat acquisition in next-generation, small area power electronics devices (e.g., processor chip, laser diodes).

Technology State of the Art: Spray cooling with a single-phase fluid is capable of heat removal at levels approaching 200 W/cm^2, but erosion and collection of spray effluent in zero gravity may be issues. Advanced heat pipes have demonstrated flux removal to ~80 W/cm^2. Two-phase thermal control systems are widely used on Earth, but lack of understanding of the effects of zero gravity has prevented widespread use in space to date for large-capacity systems.

Parameter, Value:	TRL
Heat flux: 200-400 Watts/cm^2	3

Technology Performance Goal: Provide heat acquisition and heat transfer rates that support > 1000 W/cm^2 heat fluxes in phase change heat exchangers, two-phase pumped loops, and heat pipe capillary loops for advanced power systems cooling and next-generation, high performance power electronics.

Parameter, Value:	TRL
Heat flux: > 1000 W/cm^2 Temperature differentials: < 20 K Pressure drops: < 20 kPa	7

Technology Development Dependent Upon Basic Research or Other Technology Candidate: None

CAPABILITY

Needed Capability: Thermal control systems that lower mass and accommodate ultra-high heat loads with as close to isothermal conditions as possible for advanced power systems cooling and next-generation, high-performance power electronics.

Capability Description: Provide two-phase flow systems that take advantage of the latent heat of phase change to significantly increase heat transfer per unit of system mass, as well as provide an isothermal environment at the heat exchanger/coldplate/heat transfer interfaces and stable flow conditions.

Capability State of the Art: Most spacecraft thermal control systems are single phase. Some pool boiling experiments have been successfully completed on the Shuttle and ISS. Thermal management system sizing (mass, volume) is limited by heat flux capabilities in 200-400 W/cm^2 range. Two-phase flow stability is questionable with large pressure drops.

Parameter, Value:

Heat flux: 200-400 Watts/cm^2

Capability Performance Goal: Provide heat acquisition and heat transfer rates that support > 1000 W/cm^2 heat fluxes in phase change heat exchangers, two-phase pumped loops, and heat pipe capillary loops for advanced power systems and next-generation, high performance power electronics.

Parameter, Value:

Heat flux: > 1000 W/cm^2

Temperature differentials: < 20 K

Pressure drops: < 20 kPa

Technology Needed for the Following NASA Mission Class and Design Reference Mission	Enabling or Enhancing	Mission Class Date	Launch Date	Technology Need Date	Minimum Time to Mature Technology
Exploring Other Worlds: DRM 7 Crewed to Lunar Surface	Enhancing	2027	2027	2021	3 years
Planetary Exploration: DRM 8a Crewed Mars Orbital	Enhancing	2033	--	2027	3 years
Planetary Exploration: DRM 9 Crewed Mars Surface Mission (DRA 5.0)	Enhancing	2033	--	2027	3 years

14.2 Thermal Control Systems
14.2.2 Heat Transport

14.2.2.11 Integrated Structural, Thermal, and Optical Computer Software

TECHNOLOGY

Technology Description: Integrated structural, thermal, and optical (STOP) computer software for design analyses and performance predictions of spacecraft components.

Technology Challenge: Challenges include developing full STOP analysis for instruments or spacecraft components.

Technology State of the Art: Commercial software capable of integrating the standard, discipline-specific tools used in STOP analysis have been developed in the last several years, and are currently being used at some NASA centers and a few commercial and government organizations.

Parameter, Value:	TRL
Commercial STOP analysis tool integration software distribution: Limited use distribution	5

Technology Performance Goal: Improve the capabilities of existing commercial and in-house codes that perform end-to-end STOP analyses for complex instruments and components.

Parameter, Value:	TRL
End-to-end STOP analyses integration software for complex instruments and components distribution: Widely Integrated/ Distributed	9

Technology Development Dependent Upon Basic Research or Other Technology Candidate: None

CAPABILITY

Needed Capability: Software improvements for integrated thermal-structural-optical performance analysis.

Capability Description: Enhanced, integrated STOP analysis software.

Capability State of the Art: Science missions have become more dependent on optically sensitive instruments and systems, and as such, the effects of thermal distortion on the performance of the system are critical. There are a few commercial and custom software packages that can perform integrated STOP analysis for end-to-end system performance by integrating common analysis tools. This has been demonstrated on several small-scale projects, but has lower TRL for large-scale systems.

Parameter, Value:

Integrated thermal-structural-optical performance analysis for small scale systems.

Capability Performance Goal: Improvements to existing commercial and in-house software tools that integrate commonly used discipline programs at NASA for STOP analysis. Improve speed of setup, usability, and training programs and materials for users.

Parameter, Value:

Improved usability, and capabilities for large-scale instruments.

Technology Needed for the Following NASA Mission Class and Design Reference Mission	Enabling or Enhancing	Mission Class Date	Launch Date	Technology Need Date	Minimum Time to Mature Technology
Earth Systematic Missions: Climate Absolute Radiance and Refractivity Observatory (CLARREO)	Enhancing	--	2021*	2016	2 years
Earth Systematic Missions: Aerosol-Cloud-Ecosystems (ACE)	Enhancing	--	2024*	2020	2 years

*Launch date is estimated and not in Agency Mission Planning Model (AMPM)

14.2 Thermal Control Systems
14.2.3 Heat Rejection and Energy Storage

14.2.3.1 Radiator Surface Dust Control

TECHNOLOGY

Technology Description: Specialized passive coatings (or active control) that will reduce or eliminate dust on a radiator surface.

Technology Challenge: Challenges include demonstrating long-term durability on multiple surfaces.

Technology State of the Art: Demonstrated ability to reduce dust accumulation.

Parameter, Value:
Radiator surface optical degradation.
Reduction in dust accumulation: 90%

TRL 6

Technology Performance Goal: Demonstrate the ability to reduce dust accumulation and survive in a space environment.

Parameter, Value:
Radiator surface optical degradation.
Reduction in dust accumulation: 90% and survive in space environment.

TRL 9

Technology Development Dependent Upon Basic Research or Other Technology Candidate: Materials and sensors.

CAPABILITY

Needed Capability: Reduction/elimination of dust build-up on radiator surface.

Capability Description: Provide specialized coatings or active techniques that can shed dust particles through surface energy effects, and be durable in a space environment.

Capability State of the Art: "Lotus" coating as a component of radiator paints.

Parameter, Value:
Reduction in dust accumulation.

Capability Performance Goal: Significant improvement in eliminating dust accumulation on radiator (and sliding) surfaces.

Parameter, Value:
Approx 90% improvement in dust mitigation over conventional radiator coatings, and the abillity to survive in the space environment.

Technology Needed for the Following NASA Mission Class and Design Reference Mission	Enabling or Enhancing	Mission Class Date	Launch Date	Technology Need Date	Minimum Time to Mature Technology
Into the Solar System: DRM 5 Asteroid Redirect – Crewed in DRO	Enhancing	2022	2022	2015 - 2021	4 years
Exploring Other Worlds: DRM 6 Crewed to NEA	Enhancing	2027	2027	2021	4 years
Exploring Other Worlds: DRM 7 Crewed to Lunar Surface	Enhancing	2027	2027	2021	4 years
Exploring Other Worlds: DRM 8 Crewed to Mars Moons	Enhancing	2027	2027	2021	4 years
Planetary Exploration: DRM 8a Crewed Mars Orbital	Enhancing	2033	--	2027	4 years
Planetary Exploration: DRM 9 Crewed Mars Surface Mission (DRA 5.0)	Enhancing	2033	--	2027	4 years
Planetary Exploration: DRM 9a Crewed Mars Surface Mission (Minimal)	Enhancing	2033	--	2027	4 years

14.2 Thermal Control Systems
14.2.3 Heat Rejection and Energy Storage

14.2.3.2 Two-Phase Pumped Loop System

TECHNOLOGY

Technology Description: High-capacity, two-phase heat transport systems for thermal control of large heat loads, such as those required by Rankine cycle power plants.

Technology Challenge: Challenges include validating operation of two-phase system, which requires a demonstration, and demonstrating critical heat flux in the microgravity environment as a function of flow. Additionally, heat input is required over the full range of operations.

Technology State of the Art: Two-phase thermal control systems are widely used on Earth, but lack of understanding the effects of zero gravity has prevented widespread use in space to date for large-capacity systems.

Parameter, Value:	**TRL**
Range of operation	3
Temperature control	
System life	
Mass	

Technology Performance Goal: Demonstrate two-phase thermal control in microgravity for high-power systems.

Parameter, Value:	**TRL**
Range of operation	7
Temperature control	
System life	
Mass	

Technology Development Dependent Upon Basic Research or Other Technology Candidate: Fluid boiling/evaporation and condensation phase change behavior in partial or microgravity.

CAPABILITY

Needed Capability: Thermal control systems that lower mass for high heat loads by providing isothermal conditions at the coldplate/heat exchanger side and take advantage of the latent heat of vaporization.

Capability Description: Provide two-phase flow systems that take advantage of the latent heat of phase change to significantly increase heat transfer per unit of system mass, as well as provide an isothermal environment at the heat exchanger/coldplate interface.

Capability State of the Art: Most spacecraft thermal control systems are single phase. Some pool boiling experiments have been successfully completed on the Shuttle and the International Space Station (ISS). A more realistic flow boiling and condensation experiment (FBCE) is planned on the ISS in 3 to 4 years, which will significantly advance the SOA.

Parameter, Value:

Current critical heat flux models exist with some validation on zero-gravity aircraft.

Capability Performance Goal: Two-phase flow systems such as loop heat pipes that take advantage of the latent heat of phase change to significantly increase heat transfer per unit of system mass, as well as provide an isothermal environment at the heat exchanger/coldplate interface. This technology increases in benefit with higher heat loads.

Parameter, Value:

Provide two orders of magnitude improvement (over single-phase systems) in heat transfer per unit of system mass for large-scale systems.

Technology Needed for the Following NASA Mission Class and Design Reference Mission	Enabling or Enhancing	Mission Class Date	Launch Date	Technology Need Date	Minimum Time to Mature Technology
Planetary Exploration: DRM 9 Crewed Mars Surface Mission (DRA 5.0)	Enabling	2033	--	2027	5 years
New Frontiers: Io Observer	Enabling	--	2029	2021	5 years
Discovery: Discovery 14	Enabling	--	2023	2020	5 years

*Launch date is estimated and not in Agency Mission Planning Model (AMPM)

14.2 Thermal Control Systems
14.2.3 Heat Rejection and Energy Storage

14.2.3.3 Phase Change Heat Exchanger – Phase Change Material Thermal Storage (Heat Sinks & Storage)

TECHNOLOGY

Technology Description: A phase change material (PCM) is used to store thermal energy during hot phases of cyclic thermal environments for later rejection during cold phases.

Technology Challenge: Challenges include managing the phase boundary as water freezes in the heat exchanger to prevent damage due to expansion.

Technology State of the Art: Waxes are currently used for PCM due to desirable melting point and thermal expansion during freeze/thaw. Wax provides 200 kJ/kg of PCM. Water PCMs are being developed for high heat of fusion.

Parameter, Value: Energy storage per mass of phase change material: 200 kJ/kg of PCM for wax.

TRL: 4

Technology Performance Goal: Increase energy storage per mass of phase change material to 333 kJ/kg of PCM.

Parameter, Value: Energy storage per mass of phase change material: 333 kJ/kg of PCM for water.

TRL: 7

Technology Development Dependent Upon Basic Research or Other Technology Candidate: Freeze/thaw phase change behavior in partial or microgravity.

CAPABILITY

Needed Capability: Balance spacecraft thermal load during cyclical mission environments.

Capability Description: Provide a reliable water phase change heat exchanger with 40% mass reduction over current wax PCM technology. A special case is development of PCM technology suitable for extending the life of a Venus lander.

Capability State of the Art: PCM heat exchangers provide thermal storage during relatively hot portions of missions with cyclic thermal environments. Heat can later be rejected during relatively cold portions of the cycle. PCMs currently use wax as the phase change medium. These have been used in small applications as well as on Skylab and the Lunar Rover.

Parameter, Value: Thermal energy storage per unit mass of phase change material (KJ/Kg): Current wax systems provide ~200 KJ/Kg of phase change material.

Capability Performance Goal: Demonstration of water PCM may result in energy storage increase from 200 KJ/Kg to 333 KJ/Kg. Further advances in Hx configuration may reduce containment mass.

Parameter, Value: Thermal energy storage per unit mass of phase change material (KJ/Kg): Future water-based systems may provide ~333 KJ/Kg of phase change material.

Technology Needed for the Following NASA Mission Class and Design Reference Mission	Enabling or Enhancing	Mission Class Date	Launch Date	Technology Need Date	Minimum Time to Mature Technology
Into the Solar System: DRM 5 Asteroid Redirect – Crewed in DRO	Enabling	2022	2022	2015 - 2021	2 years
New Frontiers: New Frontiers Program 4 (NF4/~2017 AO Release)	Enabling	--	2024	2016	2 years

14.2 Thermal Control Systems
14.2.3 Heat Rejection and Energy Storage

14.2.3.4 Evaporative Cooling

TECHNOLOGY

Technology Description: Evaporative cooling through water membrane evaporator.

Technology Challenge: This technology depends on water evaporation through a membrane and reaction of the resulting water vapor in an exothermic sorbent bed. The bed is sinked to a radiator that rejects heat as the water is adsorbed. The system is later regenerated with heat and gas purge. Balancing this system in a portable life support system volume is the technology challenge.

Technology State of the Art: The International Space Station (ISS) extravehicular mobility unity currently uses a water sublimator that uses approximately 3.5 kg of water per 8-hour extravehicular activity (EVA) and is highly sensitive to feed-water contamination. The Orion ammonia boiler operates during reentry and post-landing mission phases.

Parameter, Value:	TRL
Contamination-sensitive and consumes 3.5 kg/ 8-hour EVA.	5

Technology Performance Goal: 100 EVA life capability with 90% reduction in consumables.

Parameter, Value:	TRL
Minimal contamination sensitivity and 90% loop closure on evaporated water.	7

Technology Development Dependent Upon Basic Research or Other Technology Candidate: Fluid boiling/evaporation and condensation phase change behavior in partial or microgravity.

CAPABILITY

Needed Capability: Contamination-insensitive evaporative cooling with low consumables to provide heat sink for space suits and spaceflight vehicles.

Capability Description: Provide contamination-insensitive evaporative cooling for space vehicles and space suit life support systems.

Capability State of the Art: Evaporative cooling systems are currently used as thermal heat sinks for spacesuit systems (sublimators) and for space vehicle cooling during some mission phases. This includes water sublimators and ammonia boilers. Current systems consume fluids as evaporants are lost to the space vacuum environment.

Parameter, Value:

Cooling provided per mass of consumables used. Contamination insensitivity and consumable reclamation are desired.

Capability Performance Goal: Use water evaporative cooling for future space suit systems in a contamination-insensitive membrane evaporator. Close the water loop to 90% by reclaiming water after the EVA.

Parameter, Value:

Contamination sensitivity

Watts/Kg heat removal to consumable mass

Overall system mass

Consumables mass

Technology Needed for the Following NASA Mission Class and Design Reference Mission	Enabling or Enhancing	Mission Class Date	Launch Date	Technology Need Date	Minimum Time to Mature Technology
Into the Solar System: DRM 5 Asteroid Redirect – Crewed in DRO	Enhancing	2022	2022	2015 - 2021	2 years
Planetary Exploration: DRM 9 Crewed Mars Surface Mission (DRA 5.0)	Enabling	2033	--	2027	2 years
Planetary Exploration: DRM 9a Crewed Mars Surface Mission (Minimal)	Enabling	2033	--	2027	2 years

14.2 Thermal Control Systems
14.2.3 Heat Rejection and Energy Storage

14.2.3.5 Freezable/Stagnating Radiator (Variable Heat Rejection Radiator Technology)

TECHNOLOGY

Technology Description: Freezable/recoverable radiator fluid paths to allow spacecraft radiators to freeze during cold mission environments and predictably thaw for resumed operation in hot environments.

Technology Challenge: Challenges include the ability to predictably recover radiator operation from freeze/stagnation.

Technology State of the Art: Current systems use single-loop architecture with non-toxic fluid (water) with either stagnating or freeze tolerant radiator.

Parameter, Value: Demonstrated via thermal model and in laboratory 1-g environment at subscale level.

TRL: 4

Technology Performance Goal: Predictable freeze recovery from minimum mission thermal environments and low heat loads.

Parameter, Value: Stagnation/freeze recovery.

TRL: 7

Technology Development Dependent Upon Basic Research or Other Technology Candidate: Freeze/thaw phase change behavior in partial or microgravity.

CAPABILITY

Needed Capability: Radiator turn-down heat rejection.

Capability Description: Provide variable heat rejection capability across mission thermal environments and spacecraft heat loads.

Capability State of the Art: Current human spaceflight vehicles use a dual-loop architecture that allows for use of high-performance but toxic fluids on the exterior loop and benign fluids in the crew enclosure. This architecture allows use of external loop fluids that do not freeze in all mission environments. Freezable or stagnating radiators allow thermal transport fluids to freeze or stagnate in the radiator during cold mission phases with low heat loads while allowing for predictable recovery in warmer environments with higher heat loads.

Parameter, Value: Turn-down ratio in effective emissivity of radiator system: 3:1.

Capability Performance Goal: Improved turndown ratio for heat rejection of radiator system.

Parameter, Value: Turndown ratio in effective emissivity of radiator system of 6:1 (with a stretch goal of 12:1).

Technology Needed for the Following NASA Mission Class and Design Reference Mission	Enabling or Enhancing	Mission Class Date	Launch Date	Technology Need Date	Minimum Time to Mature Technology
Into the Solar System: DRM 5 Asteroid Redirect – Crewed in DRO	Enabling	2022	2022	2015 - 2021	5 years

14.2 Thermal Control Systems
14.2.3 Heat Rejection and Energy Storage

14.2.3.6 Variable-Geometry Radiators (Variable Heat Rejection Radiator Technology)

TECHNOLOGY

Technology Description: Variable geometry radiators allow heat rejection turn-down by varying the radiating surface's view to the radiative heat sink.

Technology Challenge: Challenges include implementing reliable geometry change in radiator surfaces to meet heat rejection needs.

Technology State of the Art: Current systems use dual-loop heat transport combined with body-mounted or deployable radiators. Turndown ratios can be managed to 3:1 by pointing radiators in LEO.

Parameter, Value:	**TRL**
Control of view of radiative surface to space thermal sink. Turndown ratio: 3:1 by pointing ratios in LEO	2

Technology Performance Goal: 50:1 turndown ratio of radiator heat rejection by controlling view of radiative surface to space thermal sink.

Parameter, Value:	**TRL**
Control of view of radiative surface to space thermal sink. Turndown ratio in effective emissivity of radiator system 6:1 (with a stretch goal of 12:1).	7

Technology Development Dependent Upon Basic Research or Other Technology Candidate: None

CAPABILITY

Needed Capability: Radiator turndown heat rejection.

Capability Description: Provide variable heat rejection capability across mission thermal environments and spacecraft heat loads.

Capability State of the Art: Current human spaceflight vehicles use a dual-loop architecture that allows for the use of high-performance but toxic fluids on the exterior loop and benign fluids in the crew enclosure. This architecture allows use of external loop fluids that do not freeze in all mission environments. Variable-geometry radiators allow for radiator geometry to change depending on thermal environments and heat loads. This regulates heat loss by changing the radiator's view to the space radiative heat sink and prevents transport fluids from freezing.

Parameter, Value:

Range of radiator heat rejection ($W/m^2 \cdot °K$) during mission phases with varying heat loads and thermal environments (turndown ratio).

Turndown ratio: 3:1

Capability Performance Goal: Improved turndown ratio for heat rejection.

Parameter, Value:

Range of radiator heat rejection ($W/m^2 \cdot °K$) during mission phases with varying heat loads and thermal environments (turn-down ratio).

Turndown ratio: 6:1 (with a stretch goal of 12:1).

Technology Needed for the Following NASA Mission Class and Design Reference Mission	Enabling or Enhancing	Mission Class Date	Launch Date	Technology Need Date	Minimum Time to Mature Technology
Into the Solar System: DRM 5 Asteroid Redirect – Crewed in DRO	Enabling	2022	2022	2015 - 2021	5 years

14.2 Thermal Control Systems
14.2.3 Heat Rejection and Energy Storage

14.2.3.7 Variable-Emissivity Radiator (Variable Heat Rejection Radiator Technology)

TECHNOLOGY

Technology Description: Material coatings or electrical layers that allow control of surface emissivity to manage radiated energy.

Technology Challenge: Challenges include ensuring survivability in the space environment.

Technology State of the Art: Electro-chromic or electro-static coatings that change in response to an electrical voltage potential, or passively due to temperature changes, in an Earth environment.

Parameter, Value:

Range of commanded or responsive surface emissivity.

TRL 3

Technology Performance Goal: 6:1 ratio of commanded or responsive emissivity change, and survival in a space environment.

Parameter, Value:

Range of commanded or responsive radiator surface emissivity.

TRL 9

Technology Development Dependent Upon Basic Research or Other Technology Candidate: Development of electrochemical surface coatings.

CAPABILITY

Needed Capability: Radiator turn-down heat rejection.

Capability Description: Provide variable heat rejection capability across mission thermal environments and spacecraft heat loads.

Capability State of the Art: Current human spaceflight vehicles use a dual-loop architecture that allows for use of high-performance but toxic fluids on the exterior loop and benign fluids in the crew enclosure. This architecture allows the use of external loop fluids that do not freeze in all mission environments. Robotic spacecraft also need to match radiator capacity to meet load nand prevent freezing or excessive use of electrical heaters. Variable-emissivity radiators allow for changing surface emissivity depending on thermal environments and heat loads. This regulates heat loss by reducing the surface emissivity of the radiator during cold mission phases and prevents transport fluids from freezing.

Parameter, Value:

Change in emissivity of at least 0.3.

Capability Performance Goal: Improved turndown ratio for heat rejection.

Parameter, Value:

Change in emissivity of > 0.8.

Radiator heat rejection ($W/m^2 \cdot {}^\circ K$) turn-down ratio of 6:1 (with a stretch goal of 12:1) or greater.

Technology Needed for the Following NASA Mission Class and Design Reference Mission	Enabling or Enhancing	Mission Class Date	Launch Date	Technology Need Date	Minimum Time to Mature Technology
Into the Solar System: DRM 5 Asteroid Redirect – Crewed in DRO	Enabling	2022	2022	2015 - 2021	5 years

14.2 Thermal Control Systems
14.2.3 Heat Rejection and Energy Storage

14.2.3.8 Radiator Repair

TECHNOLOGY

Technology Description: Equipment, materials, and processes required to perform repair of spacecraft radiator systems in a space environment.

Technology Challenge: Challenges include developing a repair capability for radiator panel flow tubes.

Technology State of the Art: Ability to repair/replace a limited subset of fluid line hoses and connectors via extravehicular activity (EVA).

Parameter, Value:

Repair to zero leak of four thermal transport fluid line sizes and connectors on the International Space Station (ISS). Requires EVA resources.

TRL: 3

Technology Performance Goal: Attach to radiator system to repair a leak and result in zero leaks post repair.

Parameter, Value:

Time and resource requirements to install during a mission.

Post-repair leak rate.

TRL: 7

Technology Development Dependent Upon Basic Research or Other Technology Candidate: Identifying universal repair concepts for future radiator configurations.

CAPABILITY

Needed Capability: Ability to repair radiator flow passages during a mission.

Capability Description: Maintain heat rejection capability and conserve transport fluids after damage to spacecraft radiator system.

Capability State of the Art: A radiator repair capability allows for repairing radiator transport fluid flow passages during a mission to maintain heat rejection capability and conserve transport fluids. Current external cooling loop repair technologies are limited to the fluid line repair capability on the ISS (Fluid Line repair Kit - FLRK). There is currently no radiator panel coolant line repair capability.

Parameter, Value:

Radiator repair capability during flight.

Minimal loss of capability.

Minimal crew time to repair.

Capability Performance Goal: Ability to repair any external radiator system element during a mission.

Parameter, Value:

Radiator repair capability during flight.

Minimal loss of heat rejection capability and transport fluids.

Minimal mission resources to repair.

Technology Needed for the Following NASA Mission Class and Design Reference Mission	Enabling or Enhancing	Mission Class Date	Launch Date	Technology Need Date	Minimum Time to Mature Technology
Into the Solar System: DRM 5 Asteroid Redirect – Crewed in DRO	Enabling	2022	2022	2015 - 2021	3 years
Exploring Other Worlds: DRM 6 Crewed to NEA	Enabling	2027	2027	2021	3 years
Exploring Other Worlds: DRM 7 Crewed to Lunar Surface	Enabling	2027	2027	2021	3 years
Exploring Other Worlds: DRM 8 Crewed to Mars Moons	Enabling	2027	2027	2021	3 years
Planetary Exploration: DRM 8a Crewed Mars Orbital	Enabling	2033	--	2027	3 years
Planetary Exploration: DRM 9 Crewed Mars Surface Mission (DRA 5.0)	Enabling	2033	--	2027	3 years
Planetary Exploration: DRM 9a Crewed Mars Surface Mission (Minimal)	Enabling	2033	--	2027	3 years

14.2 Thermal Control Systems
14.2.3 Heat Rejection and Energy Storage

14.2.3.9 80-250˚ C Variable Conductance Heat Pipe Radiator

TECHNOLOGY

Technology Description: Heat pipe radiator for higher-temperature heat rejection typically associated with nuclear power systems. Passive and variable heat rejection using variable conductance heat pipe technology.

Technology Challenge: Challenges include delivering a low-mass system for capability.

Technology State of the Art: Titanium/water (Ti/H_2O) variable-conductance heat pipe radiator with composite facesheet.

Parameter, Value:

Turn down ratio – Theoretically can be fully shut off to conduction only

Temperature range : 80-250 ˚C (water).

TRL

3

Technology Performance Goal: Passive, variable heat rejection radiator.

Parameter, Value:

Temperatures range: 80-250 ˚C with a high turndown ratio and specific power.

TRL

6

Technology Development Dependent Upon Basic Research or Other Technology Candidate: None

CAPABILITY

Needed Capability: Passive, variable heat rejection radiator.

Capability Description: Provide a passive, variable heat rejection radiator at temperatures between 80-250˚C with a high turndown ratio and specific power.

Capability State of the Art: No flight-qualified variable heat rejection radiator in this temperature range.

Parameter, Value:

No such capability exists.

Capability Performance Goal: Heat pipe radiator for higher temperature heat rejection typically associated with nuclear power systems.

Parameter, Value:

Passive and variable heat rejection using variable-conductance heat pipe technology with a heat rejection turn-down ratio of 30:1, a heat rejection capability to mass performance of 20W/kg, and a radiator temperature range of 80-250 °C

Technology Needed for the Following NASA Mission Class and Design Reference Mission	Enabling or Enhancing	Mission Class Date	Launch Date	Technology Need Date	Minimum Time to Mature Technology
Planetary Exploration: DRM 9 Crewed Mars Surface Mission (DRA 5.0)	Enabling	2033	--	2027	10 years

14.2 Thermal Control Systems
14.2.3 Heat Rejection and Energy Storage

14.2.3.10 Planetary Lander Multi-Phase Thermal Control

TECHNOLOGY

Technology Description: Systems that absorb energy from the spacecraft and its components and remove the energy from the system by mass transfer to the ambient environment, which may be above the spacecraft operational temperatures.

Technology Challenge: Acquisition of high temperature heat entering a spacecraft, lander, or components in order to extend equipment lifetime.

Technology State of the Art: The International Space Station (ISS) extravehicular mobility unit (EMU) currently uses a water sublimator that uses approxi)mately 3.5 kg of water per 8-hour extravehicular activity (EVA) and is highly sensitive to feed-water contamination. The Orion ammonia boiler operates during re-entry post-landing mission phases

Parameter, Value:

Contamination-sensitive and consumes 3.5 kg/ 8-hour EVA.

TRL

9

Technology Performance Goal: Provide a single-phase heat absorption system for vehicle electronics and a multi-phase heat absorption system to remove energy from the pressure vessel walls of a planetary lander. System should enable a 16 to 24 hour mission lifetime on the surface.

Parameter, Value:

Coolant and heat exchanger mass, vent system performance at Venus-like temperature and pressure.

TRL

5

Technology Development Dependent Upon Basic Research or Other Technology Candidate: Fluid boiling and evaporation behavior at high pressure and venting to the Venus surface environment.

CAPABILITY

Needed Capability: Enable a Venus (or other planetary) surface mission to survive for 16 to 24 hours which may permit human-in-the-loop science investigations.

Capability Description: An expendable coolant system that removes waste heat from electronics within a Venus or other planetary lander pressure vessel and cools the pressure vessel walls to slow down the heating up of the interior. The vaporized coolant is then vented to the atmosphere. Improved thermal protection system (TPS) insulation systems may also support this objective by retarding heat acquisition during entry and when on the surface.

Capability State of the Art: Evaporative cooling systems are currently used as thermal heat sinks for spacesuit systems (sublimators) and for space vehicle cooling during some mission phases. This includes water sublimators and ammonia boilers. Current systems consume luids as evaporants that are lost to the space vacuum environment.

Parameter, Value:

Cooling provided per mass of consumables used. Contamination sensitivity.

Capability Performance Goal: Leverage state-of-the-art evaporative cooling systems for an expendable cooling system to enable operating a lander on the Venus (or other planetary) surface for 16 to 24 hours.

Parameter, Value:

Cooling provided per unit mass of consumables. Mission lifetime of Venus lander.

Technology Needed for the Following NASA Mission Class and Design Reference Mission	Enabling or Enhancing	Mission Class Date	Launch Date	Technology Need Date	Minimum Time to Mature Technology
New Frontiers: Venus In-Situ Explorer	Enabling	--	2024	2016	2 years

14.3 Thermal Protection Systems
14.3.1 Ascent/Entry TPS

14.3.1.1 Rigid Ablative Thermal Protection Systems

TECHNOLOGY

Technology Description: Ablative materials provide thermal protection from high-speed atmospheric aerothermal entry heating loads through pyrolysis of in-depth resins and surface ablation to protect the underlying spacecraft structure.

Technology Challenge: The availability and cost of arc jet testing limits the development activity. Ground testing only simulates limited portions and limited parameters of the flight envelope, resulting in issues with ground-to-flight performance traceability. Other challenges include material sustainability for extreme environment thermal protection systems (TPS) and manufacturing TPS. An improved approach is needed to fill gaps between TPS blocks.

Technology State of the Art: Phenolic impregnated carbon ablator (PICA) is manufactured in discrete blocks of limited size. Therefore, for heat shields larger than ~1 meter in diameter, blocks must be individually attached to the heat shield substructure, requiring gaps between blocks to be filled. Monolithic Avcoat is labor intensive, requiring hand-filling of each honeycomb cell. Future manufacturing capability for carbon-phenolic heat shields is jeopardized due to lack of availability of aerospace-grade Rayon fibers and associated processes.

Parameter, Value:	TRL
Heat flux of low density ablators have been flown to 200 W/cm^2 and high density material has been flown to 30,000 W/cm^2	3

Technology Performance Goal: Demonstration of TPS material performance in relevant environment.

Parameter, Value:	TRL
1) Lower areal mass, lower cost, and reduced installation complexities with same performance as SOA materials. 2) Higher-performance materials in the heat flux gap between SOA materials that are mass efficient (heat flux range of 1000 to 7000 W/cm^2) 3) Establish new version of carbon-phenolic or suitable replacement for entry environments of > 7000 W/cm^2 4) Improve sustainability of the SOA materials, including availability, processing equipment, environmental processing issues, and capability (know-how and SME assets across generations).	6

Technology Development Dependent Upon Basic Research or Other Technology Candidate: None

CAPABILITY

Needed Capability: Improved rigid TPS ablator materials.

Capability Description: Rigid ablative TPS provides the ability to perform atmospheric entry at high velocities and operating temperatures beyond reusable TPS material limits.

Capability State of the Art: Current options include: PICA, Avcoat, SLA-561V, carbon-carbon, and carbon-phenolic, which have limitations at high velocities, dual pulse, high areal mass, and manufacturing reliability. Current material challenges include manufacturing components and integrating them into a system, typified by solutions for gaps between TPS blocks and limitation of block size. Other TPS materials, such as SLA-561V, do not have as high heat flux capability and cannot support all missions. For extreme entry conditions (>11km/sec), carbon phenolic heat shields have been used in the past. However, the ability to manufacture a heritage version of the material is limited due to diminished supply of precursor material and discontinuation of key processes.

Capability Performance Goal: Develop a material easy to manufacture at a lower cost. Understand how to fill gaps between blocks of TPS material or eliminate gaps (conformal TPS). Improve heat flux and pressure capability of TPS. Develop manufacturing capability for carbon-phenolic or develop a suitable replacement material.

CAPABILITY - CONTINUED	
Parameter, Value: SLA-561V was flown on several Mars entries at heat fluxes of 100 to 200 W/cm^2 and low stagnation pressures (< 0.3 atmospheres). PICA has been tested up to a heat flux of ~1400W/cm^2 at low to moderate pressures. Avcoat has been successfully tested up to 900 W/cm^2 at low to moderate stagnation pressures. Carbon-carbon has been tested up to 10,000 W/cm^2 for brief exposure and is considered reusable with a high-temperature exterior sealant at heat fluxes between 200 to 300 W/cm^2. Fully dense carbon-phenolic has flown in an environment up to 30,000 W/cm^2 and 7-atmosphere stagnation pressure. There is a gap in heat flux capability for the SOA materials from approximately 1000 W/cm^2 to approximately 7000 W/cm^2.	**Parameter, Value:** 40% reduction in areal mass. > 1kW/cm^2 heat flux capability. ~50% cost reduction. Dual heat pulse capability demonstration. Improved material access. Improved TPS installation issues (gaps, cracking). Need to develop, test, and certify TPS materials in the heat flux range of ~1000 to ~7000 W/cm^2. Need to reduce manufacturing cost, improve manufacturing reliability and improve availability of various material constituents. Need to improve material installation technique to improve gap filling technique or eliminate gaps (conformal TPS).

Technology Needed for the Following NASA Mission Class and Design Reference Mission	Enabling or Enhancing	Mission Class Date	Launch Date	Technology Need Date	Minimum Time to Mature Technology
Into the Solar System: DRM 5 Asteroid Redirect – Crewed in DRO	Enabling	2022	2022	2015 - 2021	5 years
Exploring Other Worlds: DRM 6 Crewed to NEA	Enabling	2027	2027	2021	5 years
Exploring Other Worlds: DRM 7 Crewed to Lunar Surface	Enabling	2027	2027	2021	5 years
Exploring Other Worlds: DRM 8 Crewed to Mars Moons	Enabling	2027	2027	2021	5 years
Planetary Exploration: DRM 8a Crewed Mars Orbital	Enabling	2033	--	2027	5 years
Planetary Exploration: DRM 9 Crewed Mars Surface Mission (DRA 5.0)	Enabling	2033	--	2027	5 years
Planetary Exploration: DRM 9a Crewed Mars Surface Mission (Minimal)	Enabling	2033	--	2027	5 years

14.3 Thermal Protection Systems
14.3.1 Ascent/Entry TPS

14.3.1.2 Obsolescence-Driven Thermal Protection System Materials

TECHNOLOGY

Technology Description: Ascent thermal protection materials, such as Shuttle-era cryo-insulation, primers, and ablators containing now-banned regulated or restricted materials like hydrochlorofluorocarbons (HCFCs) and hexavalent chromium require replacement while continuing to meet technical performance requirements.

Technology Challenge: Challenges include developing replacement materials free of banned chlorofluorocarbons (CFCs) that meet all standards for constituent materials and also meet technical performance requirements.

Technology State of the Art: Space Lauch System (SLS) core stage cryo-insulation development and qualification activities.

Parameter, Value:	TRL
Maintain current performance without the use of banned, regulated, or restricted materials and processes.	6

Technology Performance Goal: Replacement materials free of banned CFCs that meet all standards for constituent materials while meeting technical performance requirements.

Parameter, Value:	TRL
Maintain current performance without the use of banned, regulated, or restricted materials and processes that have no future potential for being regulated or restricted.	6

Technology Development Dependent Upon Basic Research or Other Technology Candidate: None

CAPABILITY

Needed Capability: Environmentally-friendly thermal protection system (TPS) materials.

Capability Description: Obsolescence-driven TPS materials provide the Agency with the ability to retain current capabilities that will be lost due to environmental regulation or compliance issues with materials that formerly provided the ability to survive atmospheric entry.

Capability State of the Art: Ongoing development and qualification of HCFC-based (3rd generation) cryo-insulation foams to replace Shuttle-era material systems and their associated processes. Proactive search and testing for 4th generation candidates that have no future potential for regulated/restricted status.

Parameter, Value:

Provides the ability to meet launch vehicle performance requirements with little to no potential impact from government regulations.

Capability Performance Goal: Environmentally friendly materials that retain capability of the SOA.

Parameter, Value:

Demonstrated technical performance of new materials equal or better than the SOA.

Technology Needed for the Following NASA Mission Class and Design Reference Mission	Enabling or Enhancing	Mission Class Date	Launch Date	Technology Need Date	Minimum Time to Mature Technology
Into the Solar System: DRM 5 Asteroid Redirect – Crewed in DRO	Enabling	2022	2022	2015 - 2021	5 years
Exploring Other Worlds: DRM 6 Crewed to NEA	Enabling	2027	2027	2021	5 years
Exploring Other Worlds: DRM 7 Crewed to Lunar Surface	Enabling	2027	2027	2021	5 years
Exploring Other Worlds: DRM 8 Crewed to Mars Moons	Enabling	2027	2027	2021	5 years
Planetary Exploration: DRM 8a Crewed Mars Orbital	Enabling	2033	--	2027	5 years
Planetary Exploration: DRM 9 Crewed Mars Surface Mission (DRA 5.0)	Enabling	2033	--	2027	5 years
Planetary Exploration: DRM 9a Crewed Mars Surface Mission (Minimal)	Enabling	2033	--	2027	5 years

14.3 Thermal Protection Systems
14.3.1 Ascent/Entry TPS

14.3.1.3 Flexible/Deployable Thermal Protection System

TECHNOLOGY

Technology Description: Flexible heat shields for deployable systems provide higher heat flux capability to support a wider range of missions.

Technology Challenge: Achieving TRL 6 will require substantial ground-based testing of systems and flight-level demonstrations, as well as some advanced material development and screening.

Technology State of the Art: IRVE-3 flight demonstration at heat flux of 12 W/cm² for an inflatable, non-ablative type system. Coupon size, ground tests for higher heat flux, non-ablative systems have been successfully demonstrated up to 60 W/cm² and ablative systems have been ground tested up to ~150 W/cm².

Parameter, Value:
Non-ablative heat flux capability currently tested to 60 W/cm². Ablative tested to ~150 W/cm².

TRL
3

Technology Performance Goal: Demonstration of thermal protection system (TPS) in relevant flight environment.

Parameter, Value:
Non-ablative heat flux to 100 W/cm²
Ablative to greater than 250 W/cm²

TRL
6

Technology Development Dependent Upon Basic Research or Other Technology Candidate: None

CAPABILITY

Needed Capability: Higher heat flux capability TPS for deployable heat shields.

Capability Description: Deployable entry systems can provide an entry drag device unconstrained by launch vehicle shroud size, which reduces the ballistic coefficient and increases the energy dissipation during atmospheric entry, thereby providing the Agency with increased capabilities in terms of mass delivered, or accessible exploration surface areas.

Capability State of the Art: Rigid TPS, which is constrained in size by launch vehicle shroud. No flexible/deployable system of applicable scale has flown in a relevant environment.

Parameter, Value:
Non-ablative heat flux capability currently tested to 60 W/cm². Ablative tested to ~150 W/cm²

Capability Performance Goal: Increased heat flux capability.

Parameter, Value:
Non-ablative heat flux to 15-100 W/cm²
Ablative heat flux to 75 to > 250 W/cm²

Technology Needed for the Following NASA Mission Class and Design Reference Mission	Enabling or Enhancing	Mission Class Date	Launch Date	Technology Need Date	Minimum Time to Mature Technology
Planetary Exploration: DRM 9 Crewed Mars Surface Mission (DRA 5.0)	Enabling	2033	--	2027	4 years
New Frontiers: New Frontiers Program 4 (NF4/~2017 AO Release)	Enabling	--	2024	2016	2 years

14.3 Thermal Protection Systems
14.3.1 Ascent/Entry TPS

14.3.1.4 In-Space Thermal Protection System Repair

TECHNOLOGY

Technology Description: Damage to space vehicle entry thermal protection systems (TPS) resulting from ascent, on-orbit micrometeoroid and orbital debris (MMOD) exposure, and damage induced through other operations may compromise the ability of the TPS to adequately protect the vehicle and crew during atmospheric entry. Suitable repair technologies are required to restore entry capability.

Technology Challenge: Challenges include developing an in-space TPS repair technology that restores TPS capability to an acceptable level for entry from design reference mission (DRM) environments and also demonstrates a shelf life greater than one year.

Technology State of the Art: STA-54 ablator material was developed to fill tile damages for the Space Shuttle program. During an on-orbit demonstration, STA-54 material was successfully dispensed into damaged TPS specimens. After on-orbit cure, the specimens were subjected to reentry testing (i.e., arc jet) on the ground with heat fluxes up to 20 W/cm^2.

Parameter, Value:

Backface temperature: < 350° F (177˚ C) for aluminum structures.

Material shelf life: ~1 year.

Material performance demonstrated in ground test environment up to 20 W/cm^2.

TRL: 6

Technology Performance Goal: Repair TPS to a level acceptable for entry. Demonstrate adequate shelf life of repair materials. Demonstrate repair in DRM entry environments.

Parameter, Value:

Repair to maintain underlying structural temperature: < 350° F (177° C) for aluminum structures and cyanate ester structures < 500° F (260° C).

Material shelf life: > 1 year for near-Earth missions and longer than mission duration for Mars and asteroid missions.

Demonstrate thermal protection for heat fluxes > 80 W/cm^2.

TRL: 3

Technology Development Dependent Upon Basic Research or Other Technology Candidate: None

CAPABILITY

Needed Capability: In-space thermal protection system repair

Capability Description: Ability to repair, in the space environment, entry vehicle thermal protection system damage caused by spacecraft debris damage, micrometeoroid damage, and orbital debris damage, which provides the Agency with the ability to add an insulative capability to the TPS (often a low-to-no fault tolerant system).

Capability State of the Art: The STA-54 ablator and the associated dispensing system were deployed for potential use during the Space Shuttle Program but an actual repair was never performed. This capability is no longer maintained. Currently, there are no U.S. human-rated vehicles in operation requiring TPS repair.

Parameter, Value:

Repair to maintain underlying structural temperature < 350° F (177° C) for aluminum structures.

Material shelf life > 1 year.

Capability Performance Goal: Performance of the uncatalyzed material in MMOD teardrop damages.

Material microgravity shelf life.

Demonstration of protection during entry heating environment.

Parameter, Value:

Repair to maintain underlying structural temperature: < 350° F (177° C) for aluminum structures and cyanate ester structures < 500° F (260° C).

Material shelf life: > 1 year for near-Earth missions and longer than mission duration for Mars and asteroid missions.

Demonstrate thermal protection for heat fluxes > 80 W/cm^2.

Technology Needed for the Following NASA Mission Class and Design Reference Mission	Enabling or Enhancing	Mission Class Date	Launch Date	Technology Need Date	Minimum Time to Mature Technology
Into the Solar System: DRM 5 Asteroid Redirect – Crewed in DRO	Enhancing	2022	2022	2015 - 2021	6 years
Exploring Other Worlds: DRM 6 Crewed to NEA	Enhancing	2027	2027	2021	6 years
Exploring Other Worlds: DRM 7 Crewed to Lunar Surface	Enhancing	2027	2027	2021	6 years
Exploring Other Worlds: DRM 8 Crewed to Mars Moons	Enhancing	2027	2027	2021	6 years
Planetary Exploration: DRM 8a Crewed Mars Orbital	Enhancing	2033	--	2027	6 years
Planetary Exploration: DRM 9 Crewed Mars Surface Mission (DRA 5.0)	Enhancing	2033	--	2027	6 years
Planetary Exploration: DRM 9a Crewed Mars Surface Mission (Minimal)	Enhancing	2033	--	2027	6 years

14.3 Thermal Protection Systems
14.3.1 Ascent/Entry TPS

14.3.1.5 Thermal Protection System Integral Health Monitoring System

TECHNOLOGY

Technology Description: A thermal protection system (TPS) with an integral health monitoring system (HMS) would reduce mission risk by detecting, locating, and evaluating impact and damage.

Technology Challenge: Challenges for detection stem from design-specific issues with signal attenuation from material properties or system complexity, and sensor placement limitations due to operating temperatures, manufacturing, or access limitations. Challenges for assessing damage are also design specific. For example, as ablative or insulative component thickness increases, the diffusive nature of conductive heat transfer makes damage assessment more difficult if using lower heat flux thermographic methods.

Technology State of the Art: Distributed, self-organized optical fiber and acoustic emission sensor network demonstrating basic impact detection and damage assessment using thermography.

Parameter, Value:	TRL
Laboratory-scale demonstrator locating impact < 2.5 inches. Basic location of damage and assessment of severity.	3

Technology Performance Goal: Full-field impact and damage assessment in a TPS environment, while being minimally invasive and negligibly contributing to TPS mass and volume.

Parameter, Value:	TRL
Vehicle-scale TPS HMS demonstration in simulated flight environment locating impact events at levels lower than those necessary to cause damage, within 6 inches, and damage assessment to within 10% of actual remaining effective thickness.	6

Technology Development Dependent Upon Basic Research or Other Technology Candidate: Sensor technology development, including operating temperature limits, power consumption, and form factor for associated systems.

CAPABILITY

Needed Capability: TPS integral HMS.

Capability Description: Self-diagnosing TPS allows the vehicle to directly or indirectly assess the operating environment and the state of a critical component of the vehicle, and take action if able and required, which provides the Agency with the ability to greatly reduce certain risks associated with the TPS (often a low-to-no fault tolerant system).

Capability State of the Art: Vehicles with a relatively small number of sensors relative to vehicle size. Mars Science Laboratory heat shield had most extensive instrumentation, with 7 pressure ports and 7 thermocouple plugs. Shuttle developmental flight instrumentation consisted of 4500 sensors with 252 TCs over orbiter surface.

Parameter, Value:

Vehicles with a relatively small number of sensors relative to vehicle size.

Capability Performance Goal: Detect impact events at levels lower than those necessary to cause damage. Perform qualitative and quantitative damage assessment.

Parameter, Value:

The impact detection goal is to detect impact events at levels lower than those necessary to cause damage, and locate within 6 inches on vehicle-scale TPS. The damage assessment goal is to perform qualitative and quantitative damage assessment (within 10% of actual remaining effective thickness) and recommend appropriate remediation if necessary.

Technology Needed for the Following NASA Mission Class and Design Reference Mission	Enabling or Enhancing	Mission Class Date	Launch Date	Technology Need Date	Minimum Time to Mature Technology
Into the Solar System: DRM 5 Asteroid Redirect – Crewed in DRO	Enhancing	2022	2022	2015 - 2021	6 years
Exploring Other Worlds: DRM 6 Crewed to NEA	Enhancing	2027	2027	2021	6 years
Exploring Other Worlds: DRM 7 Crewed to Lunar Surface	Enhancing	2027	2027	2021	6 years
Exploring Other Worlds: DRM 8 Crewed to Mars Moons	Enhancing	2027	2027	2021	6 years
Planetary Exploration: DRM 8a Crewed Mars Orbital	Enhancing	2033	--	2027	6 years
Planetary Exploration: DRM 9 Crewed Mars Surface Mission (DRA 5.0)	Enhancing	2033	--	2027	6 years
Planetary Exploration: DRM 9a Crewed Mars Surface Mission (Minimal)	Enhancing	2033	--	2027	6 years

14.3 Thermal Protection Systems
14.3.1 Ascent/Entry TPS

14.3.1.6 Self-Repairing Thermal Protection System Materials

TECHNOLOGY

Technology Description: Thermal protection systems (TPS) that can return to a virgin state without external intervention by using self-healing materials.

Technology Challenge: Proof of concept has been demonstrated in materials at temperatures below mission requirements. The challenge is to replicate material performance in high-temperature materials.

Technology State of the Art: Laboratory tests of formulations of self-healing resins (and initial work on composites with these resins) indicate it is necessary to go through a processing cycle to restore 85% of mechanical properties (tested at -30, 25, 50, 100° C).

Parameter, Value:

Modulus, compression strength after impact.

TRL

2

Technology Performance Goal: Post-event thermal and mechanical property recovery (as well as material continuity, as demonstrated by vacuum retention) to withstand typical Mars entry/Earth return conditions (temperature, heat flux, normal pressure, shear force).

Parameter, Value:

Maximum temperature: 1500 to 3000° C

Heat flux: 1-5 kW/cm^2

Pressure: ~10 atm

TRL

6

Technology Development Dependent Upon Basic Research or Other Technology Candidate: None

CAPABILITY

Needed Capability: Self-repairing TPS.

Capability Description: Self-repairing TPS allows the vehicle to recover initial functional capability after sustaining damage and therefore provides the Agency with the ability to greatly reduce risks associated with the TPS (often a low-to-no fault tolerant system).

Capability State of the Art: None. Phenolic impregnated carbon ablator (PICA) and other ablators that form the basis of common spacecraft TPS do not incorporate any self-healing materials.

Parameter, Value:

Restoration of mechanical and thermal performance, as well as material continuity as measured by vacuum retention (multiple parameters, like effective thermal conductivity of system, effective allowable strength of system).

Capability Performance Goal: Restoration of thermal and mechanical properties post impact with little or no process cycle.

Parameter, Value:

90% restoration of properties.

Technology Needed for the Following NASA Mission Class and Design Reference Mission	Enabling or Enhancing	Mission Class Date	Launch Date	Technology Need Date	Minimum Time to Mature Technology
Into the Solar System: DRM 5 Asteroid Redirect – Crewed in DRO	Enhancing	2022	2022	2015 - 2021	6 years
Exploring Other Worlds: DRM 6 Crewed to NEA	Enhancing	2027	2027	2021	6 years
Exploring Other Worlds: DRM 7 Crewed to Lunar Surface	Enhancing	2027	2027	2021	6 years
Exploring Other Worlds: DRM 8 Crewed to Mars Moons	Enhancing	2027	2027	2021	6 years
Planetary Exploration: DRM 8a Crewed Mars Orbital	Enhancing	2033	--	2027	6 years
Planetary Exploration: DRM 9 Crewed Mars Surface Mission (DRA 5.0)	Enhancing	2033	--	2027	6 years
Planetary Exploration: DRM 9a Crewed Mars Surface Mission (Minimal)	Enhancing	2033	--	2027	6 years

14.3 Thermal Protection Systems
14.3.1 Ascent/Entry TPS

14.3.1.7 Multifunctional Thermal Protection Systems

TECHNOLOGY

Technology Description: Provides increased mass efficiency by incorporating other functions like structural load carrying capacity. There are a variety of multifunctional thermal protection system (TPS) concepts that could be pursued; structurally-integrated TPS will be used as an example.

Technology Challenge: Multiple concepts have been considered and all involve a variety of fabrication challenges, including material processing compatibility, and sandwich core construction for compound curved surfaces. Scalability of concepts is also an inherent challenge. System-level challenges associated with multifunctional TPS include the modeling necessary to demonstrate design trades, as well as panel joining methods that provide mechanical and thermal load transfer. Lastly, performing non-desctuctive evaluation (NDE) on a critical component like TPS is very important, and NDE method development would be required for structurally-integrated TPS.

Technology State of the Art: Ground testing has been performed on various concepts, such as truss core, honeycomb core, and composite-wrapped insulation core. A scalability test panel was fabricated and mechanically tested.

Parameter, Value:	TRL
Repeat bending load 900 in*lb/in, edgewise compression failure at 3256 lb/in.	2

Technology Performance Goal: The primary performance goal for structurally-integrated TPS is to have the TPS carry a significant fraction of the structural load while saving significant weight over the traditional, uncoupled approach.

Parameter, Value:	TRL
Ratio of mass fraction of multifunctional TPS to uncoupled solution. Areal mass: < 4 psf Operational temperature: > 1200° C Maintenance interval: > 10 flights for reusable Impact energy absorption to be studied	6

Technology Development Dependent Upon Basic Research or Other Technology Candidate: None

CAPABILITY

Needed Capability: Multifunctional TPS. Example given is structurally-integrated TPS.

Capability Description: Multifunctional TPS could provide the ability to reduce mass fraction of the replaced (or augmented) constituent systems, thereby increasing the amount of payload mass available to the Agency.

Capability State of the Art: Current TPS (i.e. Shuttle tiles and blankets) are generally not considered structurally integrated TPS. Therefore, there is no SOA.

Parameter, Value:

None

Capability Performance Goal: To reduce overall mass of the replaced components (uncoupled structure and TPS).

Parameter, Value:

Ratio of mass fraction of multifunctional TPS to uncoupled solution.

Areal mass: < 4 psf

Operational temperature: > 1200° C

Maintenance interval: > 10 flights for reusable

Impact energy absorption to be studied.

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Exploring Other Worlds: DRM 7 Crewed to Lunar Surface	Enhancing	2027	2027	2021	6 years
Exploring Other Worlds: DRM 8 Crewed to Mars Moons	Enhancing	2027	2027	2021	6 years
Planetary Exploration: DRM 8a Crewed Mars Orbital	Enhancing	2033	--	2027	6 years
Planetary Exploration: DRM 9 Crewed Mars Surface Mission (DRA 5.0)	Enhancing	2033	--	2027	6 years
Planetary Exploration: DRM 9a Crewed Mars Surface Mission (Minimal)	Enhancing	2033	--	2027	6 years

14.3 Thermal Protection Systems
14.3.1 Ascent/Entry TPS

14.3.1.8 High Temperature Seals and Thermal Barriers

TECHNOLOGY

Technology Description: Thermal Protection System (TPS) with higher operating temperatures, improved resiliency, wear resistance, and durability resulting in increased damage tolerance at higher temperatures. Thermal barriers (1425° C to 1760° C) include advanced coatings or fibers, and associated fabrication techniques and equipment.

Technology Challenge: Challenges include developing thermal barriers with increased resiliency (925° C to 1200° C) and different materials with new fabrication techniques for pre-load devices out of single crystal or ceramic materials. Additional challenges include developing thermal barriers with higher cycles, as well as more durable coatings and/or seal materials.

Technology State of the Art: TPS materials on the Shuttle Orbiter were thicker, allowing the thermal barriers to be larger and/or installed further inboard from the outer mold line, and therefore subject to lower temperatures. Current higher temperature thermal seals and barriers utilize integral polycrystalline knitted metallic wire spring tubes and ceramic fabrics.

Parameter, Value:

Maximum operating temperature, number of thermal cycles, wear resistance.

TRL 2

Technology Performance Goal: Thermal barriers capable of operating at higher temperatures resulting from lower mass (i.e. thinner) TPS and higher reentry velocities. Thermal barriers and seals having improved resiliency, wear resistance, and durability, resulting in increased damage tolerance at higher temperatures for longer life.

Parameter, Value:

For single use, up to 1760° C operating temperature. For multi-use 1200+° C, 10s of missions.

TRL 6

Technology Development Dependent Upon Basic Research or Other Technology Candidate: None

CAPABILITY

Needed Capability: High-temperature seals and thermal barriers.

Capability Description: High-temperature seals and thermal barriers inhibit conductive and convective heat transfer through interfaces in a vehicle's TPS or between engine components to protect underlying temperature-sensitive structures.

Capability State of the Art: Thermal seals and barriers on Shuttle experienced 1425° C for short-durations during reentry; current reusable/long-duration thermal barriers remain resilient at temperatures up to 925° C for only 1-3 flight cycles and need to be replaced. Dynamic application seals can withstand 1-3 thermal cycles before excessive wear necessitates replacement. Carbon fiber rope was very successfully used in Shuttle RSRM engine nozzle in very high temperature application, but this was a non-oxidizing environment and not applicable to ascent/entry TPS without extensive coating development effort.

Parameter, Value:

Maximum operating temperature: up to 925° C

Number of thermal cycles: 1 to 3

Wear resistance

Capability Performance Goal: Thermal barriers capable of operating at higher temperatures resulting from lower mass (i.e. thinner) TPS and higher reentry velocities. Thermal barriers and seals having improved resiliency, wear resistance, and durability, resulting in increased damage tolerance at higher temperatures for longer life.

Parameter, Value:

For single use, up to 1760° C operating temperature.

For multi-use 1200+° C, 10s of missions.

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Exploring Other Worlds: DRM 7 Crewed to Lunar Surface	Enabling	2027	2027	2021	6 years
Exploring Other Worlds: DRM 8 Crewed to Mars Moons	Enabling	2027	2027	2021	6 years
Planetary Exploration: DRM 8a Crewed Mars Orbital	Enabling	2033	--	2027	6 years
Planetary Exploration: DRM 9 Crewed Mars Surface Mission (DRA 5.0)	Enabling	2033	--	2027	6 years

14.3 Thermal Protection Systems
14.3.2 Thermal Protection System Modeling and Simulation

14.3.2.1 Coupled Multi-Dimensional Flow/Material Response/ Thermal/Structural Analysis

TECHNOLOGY

Technology Description: Improved analysis techniques for thermal protection system (TPS) performance in flight and ground test environments will reduce analysis time and cost and improve the fidelity of the analysis, resulting in savings of TPS mass by reducing analysis uncertainty.

Technology Challenge: Codes will need to be evaluated against ground test in relevant flight environment and flight data. Costs and expenses associated with obtaining material data properties and test (ground and flight) results are also challenges.

Technology State of the Art: Integration of material response models into 3D thermal and structural response models and aerothermal models.

Parameter, Value: Not applicable.

TRL: 2

Technology Performance Goal: Improved analysis capability.

Parameter, Value: Not applicable.

TRL: 6

Technology Development Dependent Upon Basic Research or Other Technology Candidate: Multiscale, multiphysics, and multifidelity simulation (TA 11.3.7).

CAPABILITY

Needed Capability: Coupled multi-dimensional flow/material response/thermal/structural analysis.

Capability Description: Multi-dimensional, multi-physics analysis tools for coupled aerothermodynamic, material response, thermal, and structural solutions for TPS components to improve fidelity of analysis, which provides the Agency with the ability to either have smaller margins and lower mass TPS, and/or increased confidence and reduced risks associated with TPS.

Capability State of the Art: Use of independent codes for each discipline requiring transfer of data from code to code with mesh transfer issues and fidelity issues. Most material response models are operated as 1D models and transferred to 3D thermal response codes. Some multi-dimensional material response codes have been developed, but are not fully tested or routinely used for mission support and analysis.

Parameter, Value:
Codes used in an independent, sequential manner. The typical methodology is to run material response code in 1D and then feed discrete points to 3D thermal and structural analysis codes.

Capability Performance Goal: To improve analysis techniques, which will improve analysis fidelity, reducing uncertainty and TPS overall mass.

Parameter, Value:
Coupled analysis process resulting in 3D definition of flowfield and TPS material response.

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Planetary Exploration: DRM 9 Crewed Mars Surface Mission (DRA 5.0)	Enhancing	2033	--	2027	6 years
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14.3 Thermal Protection Systems
14.3.2 Thermal Protection System Modeling and Simulation

14.3.2.2 Shock Radiation Modeling

TECHNOLOGY

Technology Description: Analytical modeling of entry shock radiance.

Technology Challenge: The overarching challenge is reduction in uncertainty. The most significant source of uncertainty in high-velocity entry radiant heating modeling is the modeling of absorption by ablation by-products (approximately half of the uncertainty). Another significant source of uncertainty is turbulence modeling in the presence of massive ablation by-products. Additionally, aft body radiation (non-equilibrium radiation) modeling is a challenge, with the radiant heating capable of becoming larger than the convective heating like the fore body problem. Non-equilibrium chemistry modeling is also a challenge, responsible for significant radiation uncertainties for planetary entry cases. A significant challenge facing the development of this technology is the limitations in the available flight and ground test data.

Technology State of the Art: Current tools are being used at and beyond the bound of where flight data is available to validate codes. The ground test data that is available is only nearly relevant, as it does not match all of the defining parameters of the reentry environment or the vehicle geometry.

Parameter, Value:	TRL
High-speed Earth return: 35% uncertainty. Mars return: +80%/-50%. Titan, Jupiter, other locations have other uncertainty ranges.	2

Technology Performance Goal: Decrease in prediction uncertainty for all planetary destinations.

Parameter, Value:	TRL
Prediction uncertainty < 25% for all planetary destinations, given perfect knowledge of atmosphere.	6

Technology Development Dependent Upon Basic Research or Other Technology Candidate: Uncertainty Quantification and Nondeterministic Simulation Methods (TA 11.3.6) and Multiscale, Multiphysics, and Multifidelity Simulation (TA 11.3.7).

CAPABILITY

Needed Capability: Shock radiation modeling.

Capability Description: Physics-based and/or empirical modeling of heating due to shock radiance during high-velocity entry, which provides the Agency with the capability to greatly reduce the uncertainty in radiative heat load and therefore margins associated with TPS (which translates into reduced mass, increased confidence, reduced risks associated with TPS).

Capability State of the Art: SOA are the Nonequilibrium Air Radiation and Hypersonic Air Radiation Algorithm which are physics-based codes. These codes are validated with experimental data from NASA's Electric Arc Shock Tube facility and available flight data. They are coupled to the Langley Aerothermodynamic Upwind Relaxation Algorithm/Data-Parallel Line Relaxation codes for flow field solutions that provide chemical composition and temperature of flow field.

Parameter, Value:

Uncertainty is very dependent on flight case. For high-speed Earth return, uncertainty is ~35% (Mars return is +80%/-50%), absorption from ablation products is one of the key drivers (about half). The rest is due largely to turbulence modeling and precursor absorption that is not being modeled in flowfield.

Planetary entry (Jupiter, Saturn, Titan, Venus, etc.) uncertainties are large.

Capability Performance Goal: Decrease in prediction uncertainty for all planetary destinations.

Parameter, Value:

Prediction uncertainty < 25% for all planetary destinations, given perfect knowledge of atmosphere.

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Planetary Exploration: DRM 8a Crewed Mars Orbital	Enabling	2033	--	2027	6 years
Planetary Exploration: DRM 9 Crewed Mars Surface Mission (DRA 5.0)	Enabling	2033	--	2027	6 years
Planetary Exploration: DRM 9a Crewed Mars Surface Mission (Minimal)	Enabling	2033	--	2027	6 years

14.3 Thermal Protection Systems
14.3.3 Thermal Protection System Sensors and Measurement Systems

14.3.3.1 Radiometers/Spectrometers

TECHNOLOGY

Technology Description: Measure shock-layer radiation. Shock-layer radiation is a function of vehicle size and reentry velocity; the larger the vehicle, the higher the velocity and, consequently, the higher the heating. Uncertainties with shock layer radiation grow exponentially to the 9th power with increased entry velocities. At these high reentry velocities, the shock layer radiation dominates heat shield heating. Measurement of shock radiance is challenged by the extreme environment and the difficulty of positioning hardware to accurately measure shock radiance.

Technology Challenge: Challenges include development of hardware to withstand environments associated with DRMs; integration of hardware with surrounding thermal protection system (TPS) and carrier structure; data acquisition bandwidth requirements for spectrometers; instrument calibration; and data reduction/interpretation, especially when imbedded in an ablator.

Technology State of the Art: Orion Exploration Flight Test (EFT)-1 heat shield radiometers have been developed and tested in a ground-based (i.e., arc jet) facility. The EFT-1 radiometers are a combination of an optical train and a sensor. The sensor is a low technology flight-proven thermopile. The optical train is experimental and has never flown before in an ablating heat shield.

Parameter, Value:

None in relevant environment.

TRL

4

Technology Performance Goal: Measurement of entry shock radiance and spectral energy content during exo-low Earth orbit (LEO) atmospheric entry.

Parameter, Value:

Operation during exposure to exo-LEO atmospheric entry heating flux environment with an integrated radiance error: < +/-20% for the integrated system.

TRL

2

Technology Development Dependent Upon Basic Research or Other Technology Candidate: None

CAPABILITY

Needed Capability: The needed technology development is primarily the optical train for a spectrometer sensor. Increasing the bandwidth of small spectrometer sensors for flight is required.

Capability Description: Entry shock radiance measurement capability is needed to provide the Agency with the understanding necessary to design TPS with smaller margins (lower mass) and reduced risks associated with TPS for high-velocity atmospheric entry.

Capability State of the Art: Two radiometers (but no spectrometers) will be used to measure shock radiance during the multipurpose crew vehicle EFT-1 mission. As currently planned, the final opportunity to measure shock layer spectra in flight is EM-1. No measurements will be made on a crewed (EM-2) vehicle to minimize risk.

Parameter, Value:

Earth-return velocity: > 10 km/s with an integrated radiance error < +/-20% for the integrated system.

Capability Performance Goal: Measurement of entry shock radiance and spectral energy content needed for exo-LEO atmospheric entry.

Parameter, Value:

Operation during exposure to exo-LEO atmospheric entry heating flux environment with an integrated radiance error < +/-20% for the integrated system.

Technology Needed for the Following NASA Mission Class and Design Reference Mission	Enabling or Enhancing	Mission Class Date	Launch Date	Technology Need Date	Minimum Time to Mature Technology
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Planetary Exploration: DRM 9 Crewed Mars Surface Mission (DRA 5.0)	Enabling	2033	--	2027	6 years
Planetary Exploration: DRM 9a Crewed Mars Surface Mission (Minimal)	Enabling	2033	--	2027	6 years

14.3 Thermal Protection Systems
14.3.3 Thermal Protection System Sensors and Measurement Systems

14.3.3.2 High-Temperature Sensors - Wireless

TECHNOLOGY

Technology Description: Full-field sensing of a variety of parameters including, temperature, pressure, strain, and heat flux, for TPS with minimal mass and volume penalty.

Technology Challenge: Challenges include the development of high-temperature packaging and sensors, electronics, resonator materials and antennas, and communication approaches, as well as integration approaches into TPS.

Technology State of the Art: Room temperature wireless technology is commercially available. Wireless sensors have been demonstrated for radio frequency identification (RFID) systems, silicon carbide electronics, and wireless pressure sensors with limited power scavenging.

Parameter, Value:	TRL
Temperature: at least 900° C for RFID, 475° C for 22 days for silicon carbide electronics, and 475˚ C for 1 hour for wireless pressure sensor with limiated power scavenging.	3

Technology Performance Goal: Maximize the number of sensors per area, minimize weight, maximize temperature, minimize power consumption, and improve signal integrity (ability to transmit over distance, with complete dataset).

Parameter, Value:	TRL
RFID with extended wireless operation at 500°C-1000°C, with 1-5% resolution.	6

Technology Development Dependent Upon Basic Research or Other Technology Candidate: None

CAPABILITY

Needed Capability: Wireless temperature sensors that can sense and transmit temperature while at elevated temperatures (500° C and above).

Capability Description: Full-field temperature measurement improves flight safety with information on health of low fault tolerant system (above 500° C for TPS).

Capability State of the Art: Wireless temperature sensors flown on aircraft and in space, but not high-temperature wireless sensors.

Parameter, Value:

None in relevant environment. Wireless RFID sensor (900° C) with 500° C circuit (laboratory demonstrations).

Capability Performance Goal: Maximize the number of sensors per area, minimize weight, maximize temperature, minimize power consumption, and improve signal integrity (ability to transmit over distance, with complete dataset).

Parameter, Value:

RFID with extended wireless operation at 500° C-1000° C, with 1-5% resolution.

Technology Needed for the Following NASA Mission Class and Design Reference Mission	Enabling or Enhancing	Mission Class Date	Launch Date	Technology Need Date	Minimum Time to Mature Technology
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14.3 Thermal Protection Systems
14.3.3 Thermal Protection System Sensors and Measurement Systems

14.3.3.3 High-Temperature Sensors – Fiber Optic

TECHNOLOGY

Technology Description: Full-field temperature sensing using highly multiplexed fiber Bragg gratings on optical fibers.

Technology Challenge: Bragg gratings “relax” and diffuse prior to material thermal limits, causing temperature limitations prohibitive to fiber optic utilization for full-field sensing. The second temperature limit is the material limit of the fiber itself, a real limit for standard silica fibers but not a significant concern for sapphire fibers.

Technology State of the Art: New grating writing techniques have developed thermally-stable gratings in silica fibers up to 1000° C and sapphire to 2000° C. Packaging approaches are being developed. A spaceflight-ruggedized system is under development.

Parameter, Value:
See Technology SOA

TRL
3

Technology Performance Goal: Develop stable Bragg gratings with spectral response suitable for lightweight, compact lasters and robust packaging capable of platform integration and harsh environment measurments from 1000° C to 2000° C while surviving application-specific vibration and shock levels.

Parameter, Value:
See Parameter, Value under Capability section below.

TRL
6

Technology Development Dependent Upon Basic Research or Other Technology Candidate: None

CAPABILITY

Needed Capability: Full-field temperature sensing using highly multiplexed fiber Bragg grating on optical fibers exposed to enduring high temperatures (> 500° C).

Capability Description: Full-field temperature measurement improves flight safety with information on health of low-fault-tolerant system (TPS).

Capability State of the Art: Traditional thermocouples have flown on many spacecraft; wireless thermocouples have flown in lower-temperature regions of spacecraft and fiber optic sensor systems have flown on aircraft.

Parameter, Value:
Operating temperature: Fiber optic sensors flown on aircraft limited to 300° C; system limited to 40° C.

Capability Performance Goal: Thousands of sensors attached to one compact, lightweight system, capable of operating at temperatures 500-2000° C. Maximum number of sensors per area, minimum weight, minimum system volume, maximum temperature, and minimum power consumption.

Parameter, Value:
See Performance Goal Description

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14.3 Thermal Protection Systems
14.3.3 Thermal Protection System Sensors and Measurement Systems

14.3.3.4 Non-Intrusive Recession and Temperature Sensors

TECHNOLOGY

Technology Description: Recession sensors measure the amount of recession and recession rate at a specific location in the thermal protection system (TPS). Develop technology for non-intrusive measurement of both recession and temperature.

Technology Challenge: Challenges include making laboratory-demonstrated systems capable of meeting flight requirements of volume, power, and mass, with acceptable temporal and spatial resolution. Other challenges include cost and access to arc jet testing for ground testing.

Technology State of the Art: Non-intrusive measurement techniques have been developed using ultrasonic techniques and demonstrated in laboratory and limited ground test facilities. Laboratory-scale components need to be adapted for flight capability.

Parameter, Value:	TRL
Successful laboratory-scale and limited ground test demonstrations.	2

Technology Performance Goal: Transfer ground test capability to flight qualified system.

Parameter, Value:	TRL
Flight qualified system that meet can measure recession in flight TPS system as a function of time and location at resolution comparable to or better than existing intrusive systems.	6

Technology Development Dependent Upon Basic Research or Other Technology Candidate: None

CAPABILITY

Needed Capability: Non-intrusive TPS recession and temperature measurement.

Capability Description: Provide the ability to refine TPS designs for high-velocity entry, resulting in lower margins and mass.

Capability State of the Art: Use of intrusive instrumentation inserted into TPS, which can have an impact on the actual recession rate and amount of recession. No use of non-intrusive systems.

Parameter, Value:

Operating temperature: 4,000° F and no practical limit on recession for intrusive. No flight tested capability for non-intrusive.

Capability Performance Goal: Flight capable, non-intrusive measurement system.

Parameter, Value:

Measure recession in flight TPS system as a function of time and location at resolution comparable to or better than existing intrusive systems.

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